FINAL REPORT

LARGE SPACE STRUCTURE EXPERIMENTS

FOR AAP

VOLUME III

CROSSED-H INTERFEROMETER FOR

LONG WAVE RADIO ASTRONOMY

REPORT NO. GDC-DCL67-009

20 September 1967

Contract NAS8-18118

Prepared for

ADVANCED SYSTEMS OFFICE
MARSHALL SPACE FLIGHT CENTER
Huntsville, Alabama

Prepared by

CONVAIR DIVISION OF GENERAL DYNAMICS
San Diego, California

NASA Project Manager
Mr. W. T. Carey
Code R-AS-VO, NASA-MSFC
Huntsville, Alabama 35812

Convair Project Manager
Mr. J. R. Hunter
581-60, P. O. Box 1128
San Diego, California 92112

FINAL REPORT

LARGE SPACE STRUCTURE EXPERIMENTS

FOR AAP

VOLUME III

CROSSED-H INTERFEROMETER FOR

LONG WAVE RADIO ASTRONOMY

REPORT NO. GDC-DCL67-009

20 September 1967

Contract NAS8-18118

Prepared for

ADVANCED SYSTEMS OFFICE
MARSHALL SPACE FLIGHT CENTER
Huntsville, Alabama

Prepared by

CONVAIR DIVISION OF GENERAL DYNAMICS
San Diego, California

NASA Project Manager
Mr. W. T. Carey
Code R-AS-VO, NASA-MSFC
Huntsville, Alabama 35812

Convair Project Manager
Mr. J. R. Hunter
581-60, P. O. Box 1128
San Diego, California 92112

FOREWORD

The purpose of this report is to present the results of a study of "Large Space Structure Experiments for AAP" conducted by the Convair division of General Dynamics for the Marshall Space Flight Center, NASA. The study was performed during the interval 15 September 1966 to 15 September 1967, at a level of approximately $275,000.00, under Contract NAS 8-18118. The final report is published in five volumes as follows:

Volume I Technical Summary

This volume summarizes the results of the entire study.

Volume II Analysis and Evaluation of Space Structure Concepts

This volume presents the results of the analysis of the 40 space structure concepts analyzed during the first half of the study.

Volume III Crossed H Interferometer for Long Wave Radio Astronomy

This volume contains the design details of the crossed H interferometer that was one of the three concepts selected at mid-term for detailed analysis.

Volume IV Focusing X-Ray Telescope

This volume contains the design details of the focusing x-ray telescope that was one of the three concepts selected at mid-term for detailed analysis.

Volume V 100-Foot Parabolic Antenna

This volume contains the design details of the parabolic antenna that was one of the three concepts selected at mid-term for detailed analysis.

ACKNOWLEDGEMENTS

TABLE OF CONTENTS

TABLE OF CONTENTS, Contd

TABLE OF CONTENTS, Contd

SECTION 1

INTRODUCTION

1.1 **BACKGROUND.** The purpose of this study was to identify and define three large space structure experiments through which the following flight objectives could be accomplished: evaluate the role of man in the deployment, assembly, alignment, maintenance, and repair of large structures in space; evaluate the performance and behavior of large structures in space from a technology viewpoint; and provide a useful space structure that can be used to fulfill a "user oriented" requirement such as a radio astronomy antenna or solar cell array.

The logical point of departure for a study such as this is to first determine the most promising areas of science and technology that will probably require large structures in space. In viewing the potential NASA missions throughout the next decade, one can conclude that some of the more prominent requirements will evolve from the areas of astronomy, communications, and, to a lesser but still significant degree, from the requirements for solar cell arrays, micrometeoroid collectors, and magnetometers.

With regard to astronomy, regions of the electromagnetic spectrum that are of interest to astronomers begin with the very long radio waves and continue through the gamma ray region. Although not all astronomers necessarily agree on which areas of the spectrum should receive the highest priority, general concensus on those bands of particular interest and specific recommendations for future astronomy in space are found in "Space Research Directions for the Future, Part 2" (Woods Hole report). This document, together with other appropriate literature, were used to guide this study in its relation to astronomy. A few of the more important conclusions and recommendations contained in the Woods Hole report are summarized in Figure 1-1. The darkened area of the line at the top of the figure signifies those regions of the spectrum in which the atmospheric attenuation is greater than 10 db, and therefore, those regions that are essentially blacked-out from the earth's surface. In these regions, astronomical observations are completely dependent on the ability to go into space. Accordingly, the Woods Hole report has made positive recommendations for space astronomy covering the entire spectrum with the exception of that region shown as radar astronomy on the figure which, according to that report, can be satisfied by ground-based observations.

Several of the bands of interest are particularly challenging to anybody interested in large space structures. First, consider the very long wave (10 m and longer) region. Antennas designed to operate in this region have two dominant characteristics: large physical dimensions and correspondingly large allowable tolerances. The Woods Hole report recommended a "broad band antenna system leading to a 20 km aperture ultimate antenna system." Although such an antenna would not be feasible during the AAP

ELECTROMAGNETIC SPECTRUM OF INTEREST
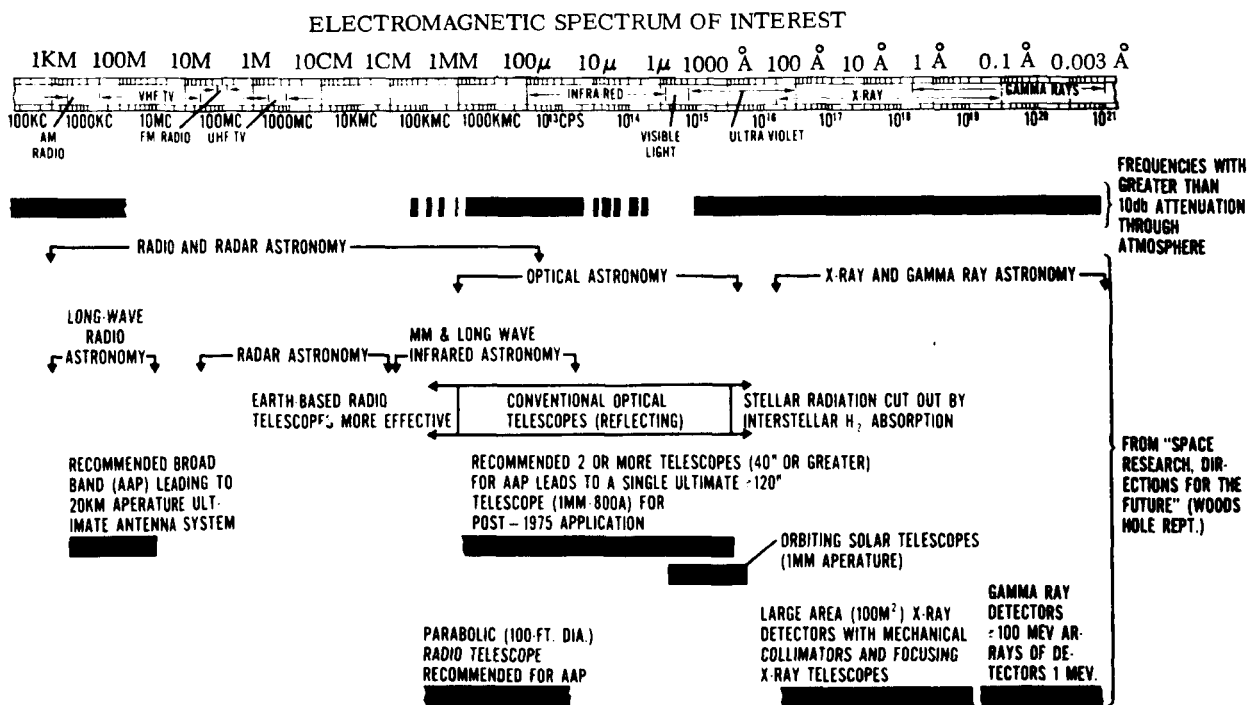
Figure 1-1. Potential Applications of Large Structures in Space

period, antenna types that may prove useful in the long wave region and, therefore, had to be accounted for in this study, include log periodics, rhombics, broadside arrays, and phased arrays. In the sub-millimeter region the primary useful antenna concept is a fairly large parabolic radio telescope with very stringent tolerance requirements. A great deal of emphasis was therefore placed in this region of interest. Additional regions of interest to astronomers, which involves large structures, are the x-ray and gamma-ray end of the spectrum. The Woods Hole report also contains specific large space structure requirements to support these astronomical programs.

In summary, Convair was directed by NASA to include in the study detailed analysis on four types of space structures relating to astronomy, longwave radio, sub-millimeter wave radio, and x- and gamma-ray astronomy. Although optical astronomy including infrared, ultraviolet, and visible regions of the spectrum are extremely important to future space flight, Convair was directed by NASA not to include these types of structures in the study.

In the area of communications many varying types of potential candidate missions exist including TV or voice broadcast type mission (of which there are numerous variations ranging from direct-to-home broadcast to simple point-to-point relay) and deep space relay. Large space structure requirements vary widely with the mission and the potential time frame for application. Although work is continuing to develop high power, long life space power supplies such as radioisotopes and nuclear reactors, there will always be a need for solar cell arrays; thus Convair was directed to include structures of this nature in the study. Additional areas that appeared to have the requirement for structures were magnetometer devices (the structural requirement emanates from the

need to separate the magnetometer a long distance from the mother spacecraft) and micrometeoroid collectors.

In summary, it was directed by NASA that the concepts to be analyzed in the study be centered around those satisfying the following user-oriented applications requirements:

> Longwave radio astronomy
> Millimeter wave radio astronomy
> X- and gamma-ray astronomy
> Communications
> Solar cell arrays
> Magnetometers
> Micrometeoroid collectors

1.2 STUDY APPROACH. Figure 1-2 shows the major tasks to be accomplished during the study as directed by NASA during the contract orientation. The study is broken into two parts. The first half deals with the analysis of a large number of candidate space structure concepts and culminates in the selection of three which then undergo more detailed analysis and design during the second half of the study.

TASK 1 10%	TASK 3 5%	TASK 4 45%
REVIEW & ANALYZE SEVERAL NASA-FURNISHED STRUCTURAL CONCEPTS PROVIDING MORE DETAILED DEFINITION OF • USER REQUIREMENTS • PERFORMANCE REQUIRE- MENTS (STRUCTURES) • STRUCTURAL DESIGN • MEANS OF DEPLOYMENT, ASSEMBLY, AND ALIGNMENT • OVER-ALL SYSTEMS REQUIREMENTS	IDENTIFY 3 ORBITAL EXPERIMENTS FOR AAP FLIGHT, THROUGH WHICH DEPLOYMENT ASSEMBLY & ALIGNMENT TECHNIQUES CAN BE EVALUATED THE PERFORMANCE OF STRUCTURAL DESIGN CAN BE EVALUATED A USER-ORIENTED REQUIREMENT CAN BE SATISFIED	PERFORM A DETAILED STUDY OF EACH EXPERIMENT PROVIDING FUNCTIONAL REQUIREMENTS CONFIGURATION DRAWINGS PACKING SYSTEMS DESIGN OPERATIONAL CONCEPT NASA FORM 1138 WORKING MODEL

TASK 2 10%		TASK 5 10%
DEVELOP ADDITIONAL CONCEPTS AS NECESSARY TO COVER A WIDE VARIETY OF FUTURE MISSION REQUIREMENTS. EACH DEVELOPED TO SAME DEPTH AS THOSE OF TASK 1		DEFINE RESEARCH, DEVELOPMENT TEST, & ENGINEERING PROGRAMS INCLUDING COSTS, REQUIRED TO FLY EACH EXPERI- MENT. PREREQUISITE ORBITAL EXPER- IMENT ALSO TO BE DEFINED (1138)

Figure 1-2. Task Areas

1.3 SUMMARY OF RESULTS. Tasks 1 and 2 resulted in the preliminary design and analysis of 40 candidate space structure concepts for flight in the 1970-75 time frame. Three of these structures were selected at the mid-term point of the study and were the subject of detailed preliminary design and analysis during the second half of the study. They are: a long wave radio astronomy antenna called a crossed-H interferometer, a focusing x-ray telescope, and a 100-foot-aperture parabolic antenna. An overall summary of the entire study can be found in Volume I. Analysis of the 40 candidate structures can be found in Volume II, and detail analysis of the three selected concepts are in Volumes III, IV, and V respectively as listed above.

1.4 INTRODUCTION TO VOLUME III. The approach utilized in the preliminary design and analysis of the crossed-H interferometer is depicted in Figure 1-3. The design includes a preliminary definition of the structure, facility subsystems, and instrumentation. A hypothetical mission was developed. The analysis has included thermodynamics, dynamics, stress, mass properties, and reliability. A functional and time-line analysis was also performed.

This volume describes the design of a large orbiting structure that fulfills a highly desirable scientific mission. The design of the instrument is the result of several meetings with members of the scientific community, thus assuring the validity of the basic design.

The resulting structural concept provides an excellent facility that will demonstrate the performance and operation of extendible boom structures and extendible tubular mesh structures under various dynamic and thermal constraints. Such evaluations as tethered body behavior and thermal environment will also be fulfilled.

The crossed-H antenna has been evaluated to determine how man may be utilized in assuring the highest probability of mission success. His primary roles are to serve as a deployment and checkout monitor and as an orbital maintenance and refurbishment medium. By the use of photographic records and medical sensors, the astronaut's dexterity and physical performance may be recorded to validate his ability to function as a valuable asset to an orbiting structure.

A NASA 1346 form for the crossed-H interferometer is included in the appendix. Section 7 presents a preliminary research, development, test, and engineering plan. A 1/10 scale model has been fabricated and, together with a movie demonstrating the deployment, delivered to the NASA-COR.
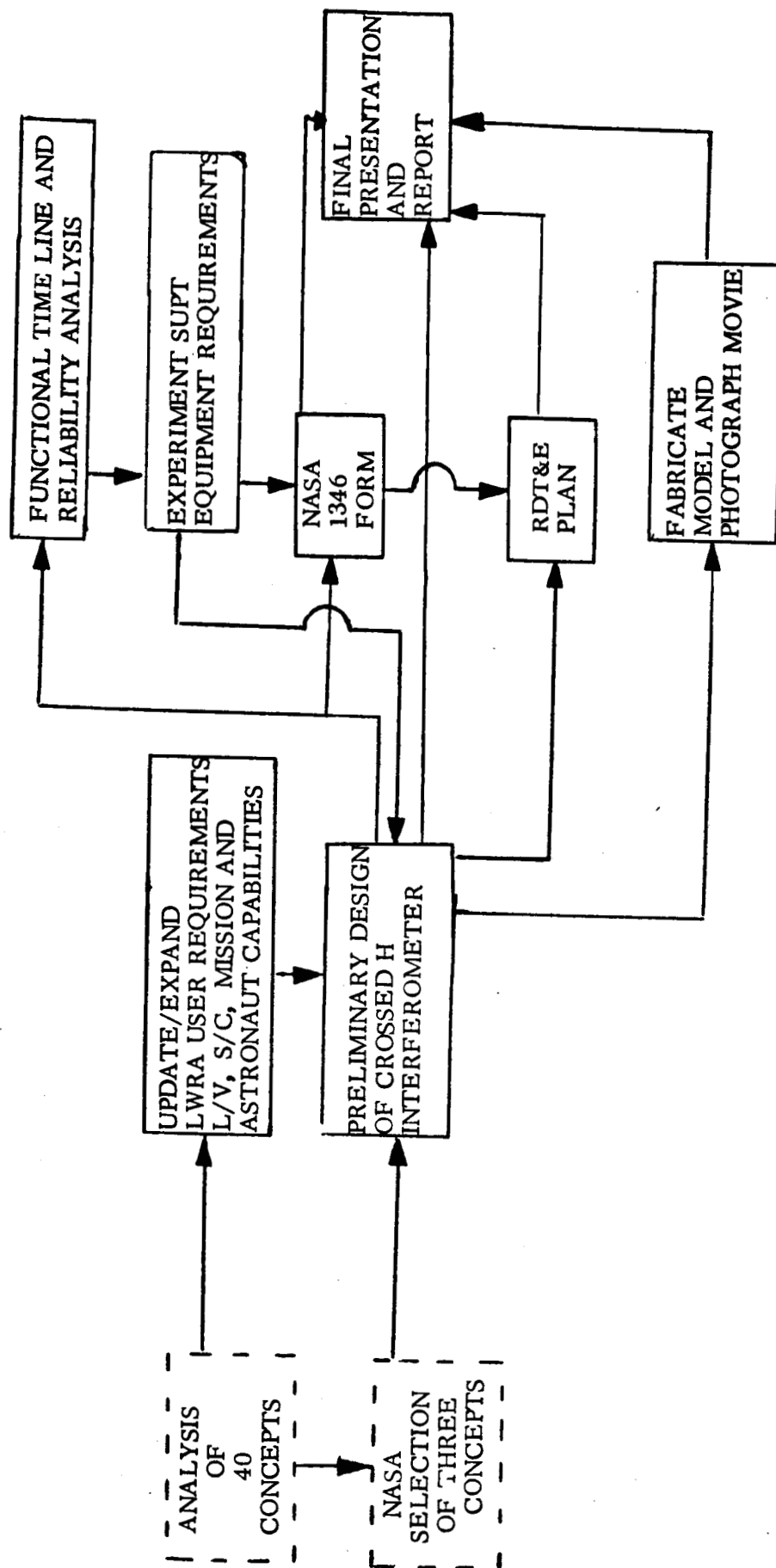
Figure 1-3. Technical Approach Crossed-H Interferometer

SECTION 2

FLIGHT OBJECTIVES

2.1 <u>MAN'S ROLE AND CAPABILITIES</u>. One of the primary flight objectives of the crossed-H antenna is to verify the use of man in erecting and maintaining the operation of a large orbiting structure.

In order to achieve this goal on a justifiable basis, the astronaut's capabilities were weighed against automation on the system level and redundancy on the component part level. Such factors as reliability, work hazards, costs and EVA work constraints were used as a basis for analysis. This analysis indicated that man's activities were indispensable in assuring mission success in the areas of:

a. Initial deployment and checkout.

b. Malfunction repair.

c. Scheduled refurbishment.

While accomplishing the above tasks, the astronaut will furnish valuable data for future orbital work through the media of photography and biomedical sensors.

Through the use of photography, the astronaut's abilities to do the assigned tasks may be recorded. His ability to handle and replace various size components from the fairly large solar cell panels to the small attitude control jet modules may be confirmed. His methods of locomotion will also furnish valuable data upon which future activity may be based. Such equipment transportation media as the "clothesline" principle may be observed for future practicality.

Biomedical sensors will provide valuable information as to the validity of assumed timespans for task accomplishment, thus providing a valid basis for required rest periods. Since the astronaut is scheduled to do a wide variety of tasks, the recorded data will also provide an excellent catalog of his physical reaction to the various tasks.

2.2 <u>STRUCTURAL TECHNOLOGY</u>. The advancement of large orbiting structural technology is one of the prime flight objectives of the crossed-H antenna. Through the media of thermal sensors, strain gauges, and photography, the structural behavior may be telemetered to earth for analysis and subsequent structural advancement.

A most significant area of technological development is the tether. Studies have been performed on tether dynamics and it is anticipated that experience with orbital tethers will be gained before the launch of this experiment. But it is probable that the use of this tether length in repeated extension and retraction will, with suitable instrumentation, provide valuable data on such a typical space structure.

The extendible boom is another example of the application of structural technology to space environment. Normal webbed truss theory is extended to provide a boom with a high degree of immunity to thermal and dynamic distortions while incorporating an approximate 8 to 1 repeatable extension. Strain gauges and thermocouples and position sensors will provide data that evaluate the effectiveness of the design and point to new improvements.

Other features of the antenna assembly provide sources of data that will establish and promote structural technology. The dipoles of bi-metallic mesh in the shape of extendible and retractable tubes should provide additional data in this application, although we anticipate that the fundamental concepts will be proven in preceding experiments or pre-orbital tests.

The various drive mechanisms are in accord with normal principles of operation on the earth, but will be further tested in space by this experiment, particularly with respect to longevity and reliability. The need for such proof is reflected in the incorporation of the detail design of redundancy and provisions for EVA replacement by means of modules and gross attachments.

In short, this design represents a logical and practical extension of structural technology to space applications. It also provides the means of evaluating such technology extension and opens the door to further progress.

2.3 SCIENTIFIC OBJECTIVES. The scientific objectives that the antenna has been designed to fulfill are:

a. Survey the VLF sky radiation over the entire sky, with good resolution, including spectral and polarization measurements.

b. Survey VLF discrete radio sources, with good resolution, including spectral and polarization measurements.

c. Obtain spectral and polarization measurements of the sun, with good temporal resolution.

d. Obtain VLF observations of Jupiter, and possibly other planetary sources, with good temporal resolution.

It is the intent in the crossed-H interferometer design to use aperture synthesis techniques to perform the investigations under "a" and "b" above; "c" and "d" may be accomplished using each end at a different spectral range. The latter is by virtue of the fact that each end separately has good front-to-back ratio of response, thus eliminating radio frequency interference from other sources, coupled with adequately narrow beamwidth. In addition, each end incorporates complete polarization measurements capability. Thus, "c" and "d" above are carried out in the frequency range from 0.5 MHz to 5 MHz, or 2.5 MHz to 10.0 MHz, simultaneously and in both polarization orientation simultaneously.

Before going into the requirements on the antenna design, it is worthwhile to consider the origin of the four objectives listed above.

2.3.1 <u>Radio Sky Brightness</u>. We are considering a region of the radio frequency spectrum where the earth's ionosphere acts as a shield to prevent ground-based observations of radio emissions. This region of the spectrum extends from frequencies of 5-10 MHz downward to a lower frequency limit, where observations prove to be impossible because of the effects of the interplanetary plasma.

The upper-frequency limit of this blocked spectral region varies as the electron density in the ionosphere varies (See Figure 2-1). Plasma frequency f_0 in kHz is approximately equal to nine times the square root of the electron density in particles per cm^3. We must have an orbital altitude during our observations such that f_0 is less than or equal to one half of our lowest observing frequency, i.e.:

$$f_o \cong 9 N_e^{1/2}$$

$$\text{Obs. freq.} \geq 2.0 f_o$$

$$\geq 18 N_e^{1/2}$$

where N_e is the number of electrons per cm^3.

Occasional observations may be made from earth at frequencies as low as 1 MHz, as Grote Reber and G. R. Ellis have shown (Reference 2). Consistent ground-based observations on frequencies below 10 MHz are, however, most difficult to achieve, partly because the critical frequency of the F region often exceeds this value and partly because, even on those occasions when radiation can penetrate the F region from directly above, long distance oblique incidence propation is possible, making ground-based radio transmitters able to send strong interfering signals to a radio-astronomy antenna and prevent radio-astronomical observations from being made.

Above the F-region ionization maximum, however, the electron density steadily falls, merging eventually with that of the plasma surrounding the sun (Reference 1). In this region of space, long-wave radio astronomy is practical. An antenna placed here can both receive signals from outside the earth and, by the shielding of the ionosphere which is now between it and the ground, be freed from interference by man-made signals generated on earth. Based on Figure 2-1, we have no problem with plasma frequency, since our orbital altitude is on the order of five earth radii.
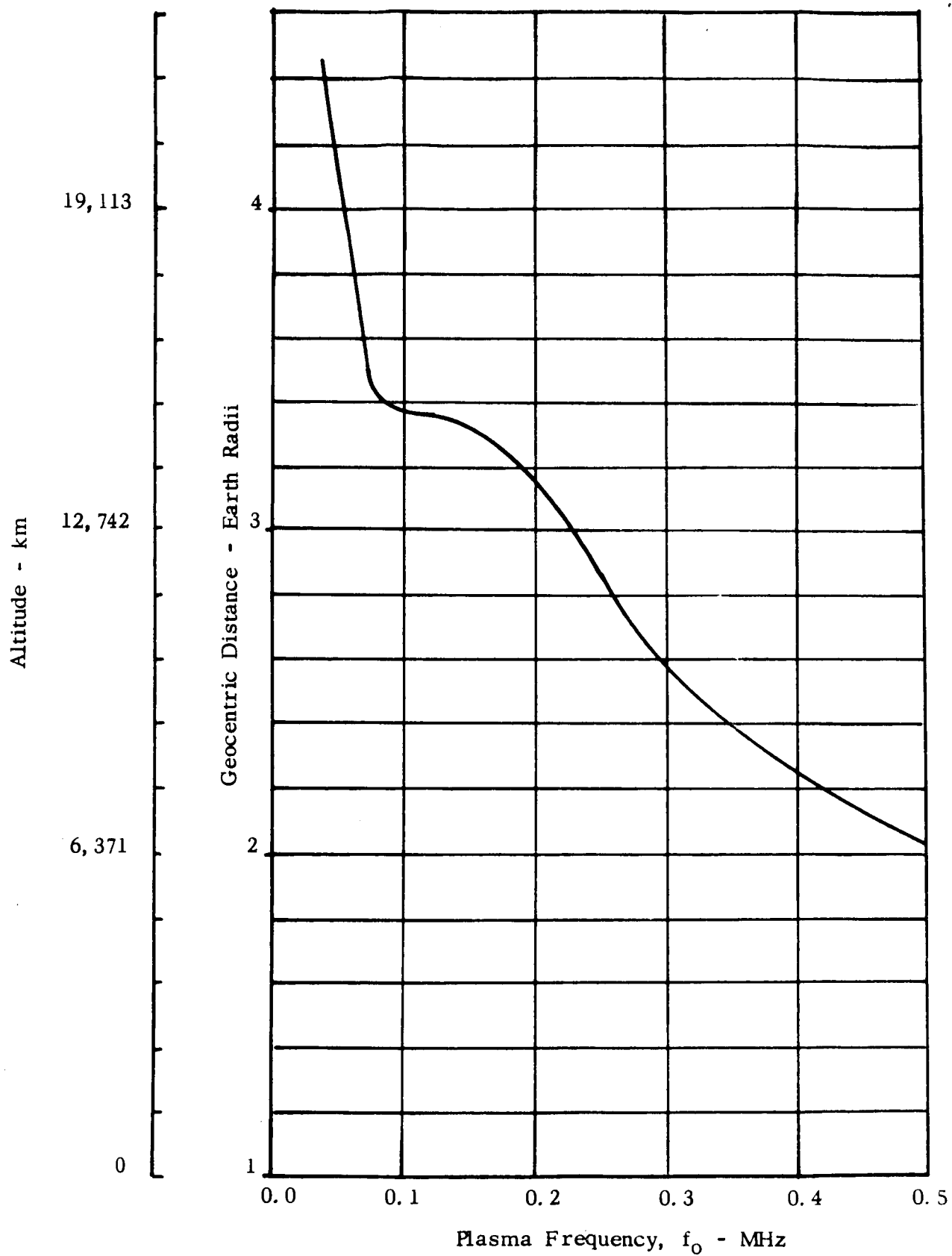
Figure 2-1. Plasma Frequency as a Function of Altitude, Taken from the Equatorial Profile of Magnetospheric Electron Density (Reference 1)

According to Haddock (Reference 3), from low-frequency observations with orbiting radio telescopes operating in the band around 1 MHz, we expect to find that the radio sky will show the following:

a. A very bright overall glow.

b. A wide dark band distributed along the galactic equator, becoming very wide toward the galactic center.

c. A large number of bright sources of emission distributed over the sky, but because of strong absorption by interstellar gas, rather sparse in the Milky Way.

d. Frequent strong outbursts of radio waves from Jupiter.

e. Occasional radio outbursts and noise storms from the sun. The undisturbed sun and most of the planets will be inconspicuous.

However, at the low-frequency end of the radio window (5 or 10 MHz), the prominent features in the sky are as follows according to Haddock:

a. Sporadic bursts of emission from the sun and Jupiter.

b. A bright belt of radio emission in the galactic plane.

c. An overall glow from the entire sky.

d. Scattered bright regions of emission from distant radio galaxies.

e. Small quasi-stellar objects of extreme radio brightness.

f. A number of disk-like bright sources varying in size and concentrated along the galactic plane, identified with remnants of the supernova.

g. Dark interstellar clouds of ionized gas, which absorb the general background emission; these also vary in size and are concentrated in the galactic plane.

As far as galactic background is concerned, all radio-astronomy experiments that have been flown in satellites have tried to measure the average brightness of the radio sky at a few places in the spectral range from 10 MHz to 725 kHz (Reference 3). Such experiments can be made with essentially nondirectional antennas and are thus relatively simple to perform. Yet the scientific results are needed to understand the mechanisms of emission and absorption of radio waves within the galaxy. The spectrum of this galactic background radiation shows a continuous increase in sky brightners as the frequency is reduced until about 2-3 MHz is reached (See Figure 2-2). One group of experimenters believes in a rather drastic decrease in intensity between 3 and 1 MHz. Another group sees a leveling off in this region. Yet another opinion believes in the possibility of an increase.
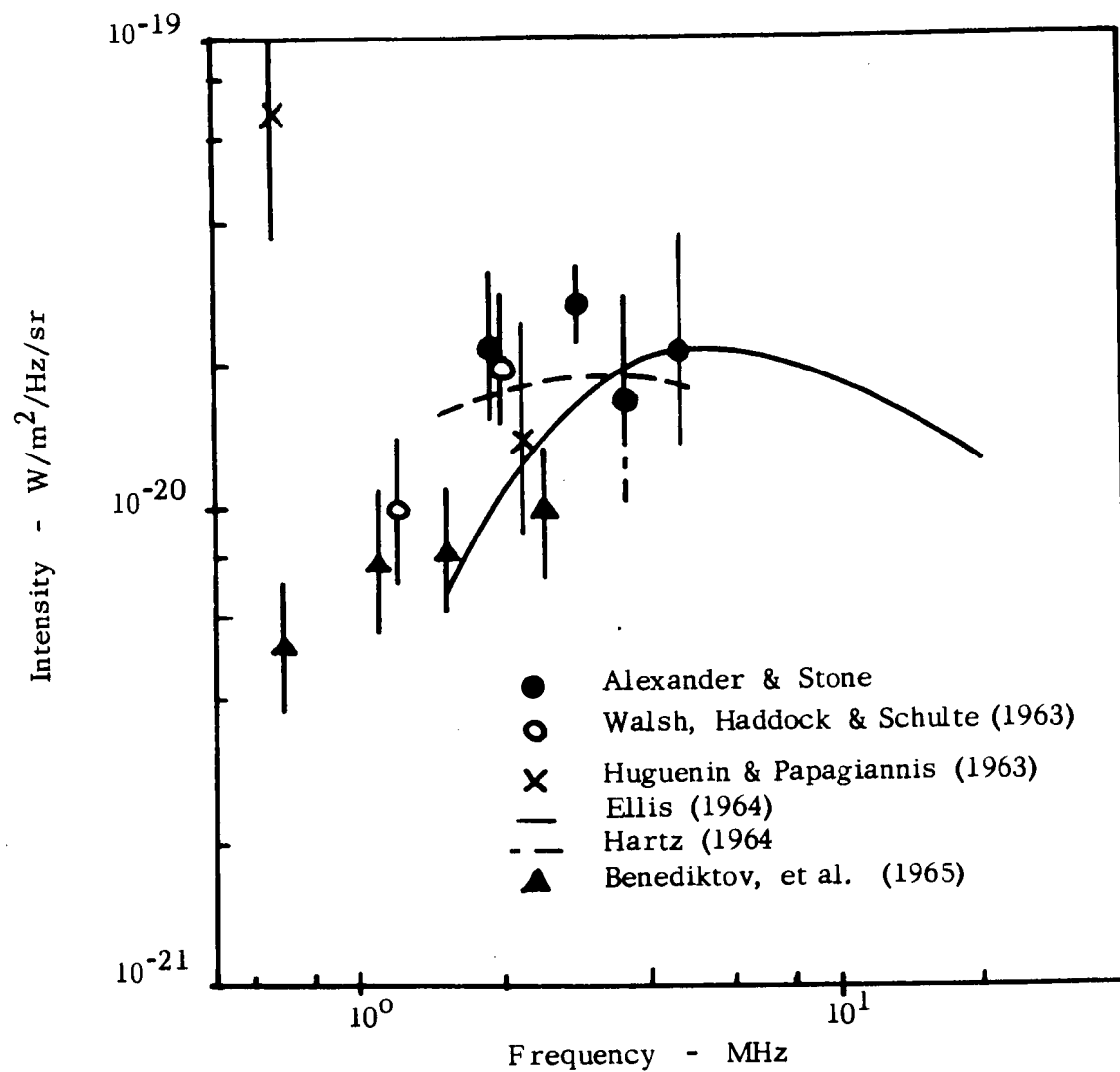
Figure 2-2. Observed Cosmic Noise Intensities Below 10 MHz (Reference 1)

The intensity values reported by Benediktov (et al) at 0.7 MHz are an order of magnitude smaller than the values reported by Huguenin (et al). Perhaps what is being sampled is earth noise. This might be expected to possess a large order of magnitude variations, whereas the cosmic background would not. Measurements made outside of the magnetosphere will perhaps resolve this enigma and give a true reading of the intensity of cosmic radio emission below 2 MHz.

The shape of the spectrum of the galactic continuum at low frequencies has consequences on other lines of endeavor. If the intensity of the emission levels off or increases, it will be difficult to detect other sources above this interference. On the other hand, if the emission falls off, the detection of weak signals in this part of the spectrum will be possible. Whatever the shape, the spectrum definition will provide input to questions of energetics, populations of cosmic ray electrons, and accelerating mechanisms. The spatial distribution of galactic radiation at low frequencies will undoubtedly lead to a clearer delineation of the spiral structure of the Milky Way.

With the improved resolution of the crossed-H interferometer we will be able to survey the VLF sky radiation to resolve some of the above uncertainties. It will permit a detailed study of the distribution and properties of ionized hydrogen in the galaxy, providing that the darkening of the Milky Way with decreasing frequency actually is found to be true. This darkening would probably be attributed to the effect of absorption due to ionized hydrogen.

2.3.2 <u>Solar and Planetary Low Frequency Radio Astronomy</u>. Another goal is the observation of radio waves emitted by the sun and the planet Jupiter (Reference 3). Both of these are remarkable and sometimes energetic radio sources. Although radio waves from the sun have been studied for 20 years and a wide variety of bursts observed and classified, the origins of these phenomena are still far from fully understood. Equally, the relations between these kinds of solar activity and terrestrial effects still present many problems. Study of the low-frequency solar bursts at these frequencies, which originate up to several solar radii out from the sun, will give information on the solar plasma. The radio waves from the undisturbed sun also require study in this spectral range.

Jupiter is the source of large radio bursts, already well observed from the ground in the frequency range from about 50 MHz to the earth's ionospheric limit near 10 MHz. The planet also emits, probably because of a radiation belt system, strong nonthermal radio radiation at microwave wavelengths (Reference 1).

The currently accessible spectral range within which the bursts may be seen is clearly limited by the earth's ionosphere at the lower-frequency end. An extension of observations here will help in understanding how the bursts originate and perhaps also in showing whether Jupiter itself is subject to the same sort of sun-induced effects as the earth.

The radio spectrum of the sun is one of enormous complexity, with at least five different types of transient outbursts in addition to the steady thermal emission. Study of these outbursts has increased our knowledge of the solar atmosphere and in particular, has led to a better understanding of centers of activity. Figure 2-3 shows the dynamic spectra of solar radio bursts - electromagnetic phenomena associated with chromospheric flares.
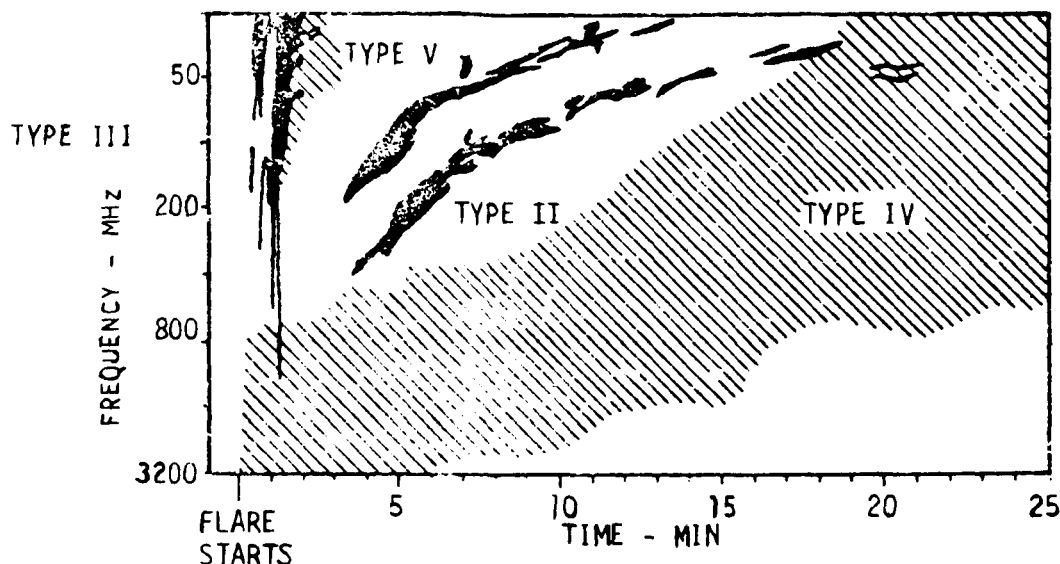
Figure 2-3. Dynamic Spectra of Solar Radio Bursts (Piddington) (Reference 1)

Since the solar radio spectrum is continuously changing, attempts to extend measurements into the low frequency region that borders the top of Figure 2-3 should be designed to fully reveal the characteristics of the bursts in the frequency-time domain. The experiment should be capable of tuning over a wide range of frequencies in a fraction of a second and recording the intensity of the radiation at each frequency. This is precisely one of the purposes the crossed-H interferometer has been designed to fulfill.

In addition to the sun, Jupiter is the only confirmed nonthermal source of radio frequency emission in the solar system. The sporadic bursts at decameter wavelengths are strongly correlated with Jovian longitude and with the position of the satellite Io, indicating that the sources of radiation are linked to the disk of the planet. Whether the mechanism responsible is a surface phenomenon or occurs in the exosphere and is coupled to the planet through its magnetic field has not yet been determined. Another nonthermal component of emission, which occurs in the microwave part of the spectrum, is believed to originate in Jovian Van Allen belts where trapped electrons radiate by the synchrotron process.

2-8

Figure 2-4 shows the spectral distribution of the average planet-wide flux density from Jupiter. The spectral index varies between -5 and -8 over much of the observable decametric range, suggesting that the bulk of the radiation may actually lie in the low-frequency tail, which is unobservable from the ground because of ionospheric cutoff. Furthermore, the apparent structure of the Jovian radio sources changes radically at frequencies below 15 MHz, and the pulse structure of the emission is unquestionably altered by the terrestrial ionosphere.

Observations of the giant planet at frequencies below 5 MHz could establish the spectral distribution of the emission; i.e., extend the curve of Figure 2-4 to the left. Such information is vital to theories concerning the origin of the radiation, total energy involved, particle densities required, and efficiency of the conversion mechanism. An examination of the pulse structure, unaltered by the terrestrial ionosphere, could yield information about Jupiter's ionosphere. Dynamic spectra (frequency versus time plots) and polarization studies at these low frequencies might unravel the mysteries of the planet's magnetic field. Long-term monitoring would undoubtedly yield evidence of propagation effects in interplanetary space, correlation with sunspot activity, and increase our knowledge of another magnetospheric system, perhaps much like our own but on a larger scale.

There have been reports of possible emissions from Saturn at low frequencies. These do not, at present, constitute hard evidence. Since modern theories tend to associate a planet's magnetic field with its rate of rotation, Saturn might be expected to emit because of its similarity to Jupiter in this respect. The fact that no confirmed nonthermal radiation from the ringed planet has been detected could be the fault of listening at the wrong frequencies. Here again, monitoring at low frequencies, unhindered by an ionosphere and atmospheric and man-made interference, could solve the riddle.

With respect to the earth, the Van Allen radiation zones probably provide magnetic bremsstrahlung emissions which could be detected from orbit. Occultation experiments with the aim of detecting propagation effects caused by the interposition of the earth's ionosphere or magnetosphere between the source and the receiver can yield valuable data on the terrestrial environment.

2.3.3 Galactic and Extragalactic Low Frequency Radio Astronomy. There are numerous discrete sources on the celestial sphere, and these have been dubbed "radio stars" because of their small angular extent. In most cases they are not stars, but nebulae or galaxies, and their emission is continuous in time (Reference 2). They can be classed as galactic or extragalactic, thermal or nonthermal. Some of the discrete sources of our own galaxy exhibit typical thermal emission spectra. These sources (galactic, thermal) are thought to be clouds of ionized gas - the coronas surrounding bright, hot stars in the spiral arms, and are referred to as H-II sources. A typical example is the source Cygnus X. All of the galactic, nonthermal sources are believed to be remnants of super novae - catastrophic explosions of stars. The Crab Nebula is an example and emission is by the synchrotron process. All of the extragalactic discrete sources, which
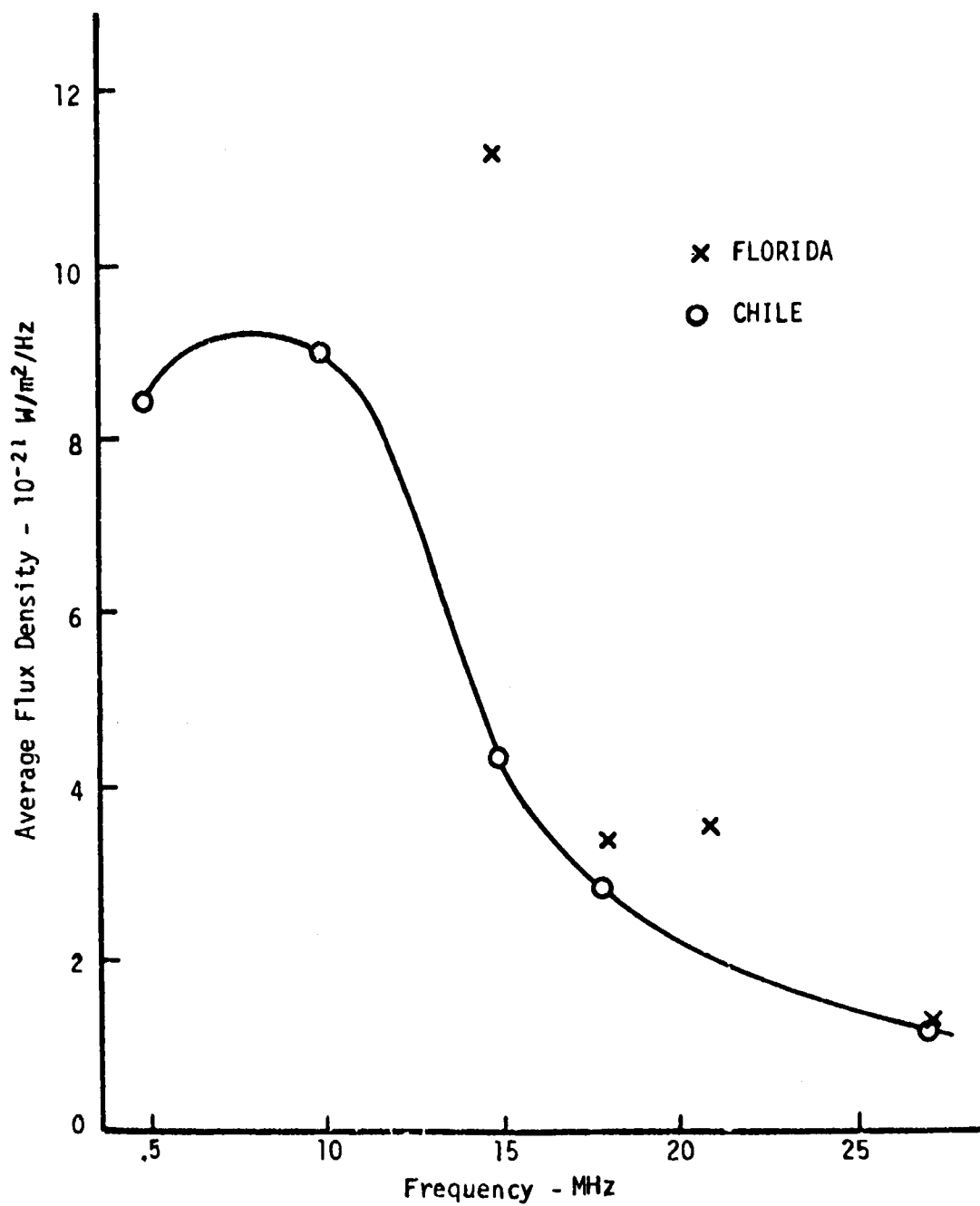
Figure 2-4. Spectral Distribution of the Average Planet-Wide Flux
Density from Jupiter (Reference 1)

2-10

constitute the vast majority of radio stars, are characterized by nonthermal spectra. Again, the synchrotron mechanism offers a reasonable explanation of this emission. These extragalactic sources, which have an isotropic distribution in space, are classified as "normal" and "peculiar". The normal sources are bright, spiral galaxies with radio outputs of approximately 10^{28} kW, i.e., the same total power as the Milky Way. The classic example is the Great Nebula in Andromeda. The peculiar sources are much more intense. For example, the source Cygnus A, believed to be two galaxies in collision, has a radio output of 10^{34} kW (Reference 1).

All of the discrete radio sources emit a fairly smooth continuum of radio frequency energy. Measurements of the intensity of these sources at low frequencies will provide additional spectral evidence that has a bearing on the stability of the emission mechanism and on the energy spectrum of the particles responsible. Surveys at frequencies below 5 MHz showing the positions in space of the discrete sources would be valuable additions to the information now in hand. This mapping, along with individual source studies to ascertain angular dimensions and intensity distributions, will require high resolution and narrow beamwidth. Such investigations are vital to attempts at optical identification of the radio sources.

The sky background will be mapped with resolution adequate to see the distribution of ionized hydrogen in interstellar space. Measurement of its absorption may lead to a new insight into the mechanisms of star formation. The contributions to the sky background from our own galaxy and from the extragalactic medium can be separated. Polarization studies will improve knowledge of the magnetic fields in interstellar space although, possibly, large Faraday rotation effects may make the results difficult to interpret.

As soon as the stronger of our known radio sources, among which are the supernova remnants, the strong radio galaxies, and the quasi-stellar sources (quasars), can be detected at lower frequencies and isolated from the galactic background, many fruitful researches should emerge.

The fluxes of these sources must be measured and their known spectra extended. In this low-frequency region it is possible that the radio-wave absorption, which occurs perhaps in or near the source or within our galaxy, can be studied and the effects of self- and galactic-absorption separated. Detailed data on the low-frequency spectra can give knowledge of one or more of the following characteristics of quasars and radio galaxies: the density of cosmic ray electrons, the magnetic field strength, the plasma density, and the low energy cutoff of the cosmic ray electrons. In both these sources unknown yet highly efficient mechanisms of energy conversion are at work.

At high frequencies, measurements of polarization in sources already suggest the presence of a galactic or a source magnetic field. There is also some evidenct that the radio flux from quasars varies with time. These phenomena should be studied at low frequencies also. These investigations are possible with the fine resoltuion achievable by the crossed-H interferometer.

SECTION 3

SCIENTIFIC REQUIREMENTS

In order to have large space structures that would fulfill the third main objective of the study, namely that of providing a scientifically useful device in earth orbit, an extensive investigation was undertaken in the first half of the study to come up with scientific user requirements. These scientific user requirements were then utilized both as evaluation criteria for the NASA supplied concepts and as design goals for driving out those new concepts that were eventually selected by NASA for further detail design study during the latter half of the study. Several prominent members of the scientific community were instrumental in devising these design goals. Convair would like to acknowledge the invaluable assistance, at no cost to NASA or Convair, of the following members of the scientific community who were concerned with long wave radio astronomy:

Dr. Tom A. Clark, Marshall Space Flight Center
Dr. William C. Erickson, University of Maryland
Dr. John P. Hagen, Pennsylvania State University
Dr. G. Richard Huguenin, Harvard Space Radio Laboratory
Dr. George R. Lebo, University of Florida
Dr. William J. Lindsay, University of Michigan
Dr. N. Frank Six, Jr., University of Western Kentucky
Dr. Robert G. Stone, Goddard Space Flight Center

3.1 <u>LONG WAVE RADIO ASTRONOMY USER REQUIREMENTS</u>. Most radio astronomy work that could be accomplished in the 1968-1974 time period is not sensitivity limited, but resolution limited (Reference 4). The general recommendation is to achieve some type of antenna interferometer array with a 2° resolution. For solar and planetary astronomy far less resolving power is necessary, since these sources are expected to be very strong in comparison to the general noise background. Good resolution will be welcomed by the astronomy community, since to date relatively poor resolution has been obtained with the dipole elements flown by Haddock, Huguenin, and others. The Radio Astronomy Explorer to be flown by Stone in 1967 will not provide significant resolving power. About 9° to 10° beamwidth seems to be accepted as adequate by the astronomers at 1 MHz providing that some interferometer arrangement could also be provided for additional resolution. The bandwidth should be as large as possible, with primary emphasis on the 0.5 - 10.0 MHz range. A capability to make polarization measurements should be provided. Specifically, the following parameters were considered:

Life Time: Minimum of 1 year desired.

Orbit Altitude: Minimum of synchronous.

Effective Beamwidth:	One-hundred deg^2 at 1 MHz desired — less than 10° in one direction, but could be greater for solar and planetary astronomy. Interferometers should be used if possible for improving this resolution to 2°.
Pointing Accuracy:	One-half half-power beamwidth or better. 1/10 for for aspect determinations. In the case of a sweeping mode or drift mode antenna system, the pointing direction must be known to within 1/10 half-power beamwidth or better.
Pointing Stability:	Approximately 1/10 beamwidth or better.
Bandwidth:	Five hundred kHz to 10 MHz desired, with emphasis on lower half. Possible extension to 200 kHz.
Spectral Resolution:	Good desired, and depends only on electronics for any one antenna.
Sensitivity:	Unfilled apertures entirely adequate.
Lock-on-Time:	One-half second to several hours for time varying phenomena. For mapping observations, however, an antenna arrangement with as slow a drift-rate as possible — of up to approximately 1 deg/min suffices.
Tolerance:	Prefer $1/20\ \lambda$, but $1/16\ \lambda$ is adequate. (At 1 MHz, $\lambda = 300$ m.)
Orientation Discrimination:	Eliminate antenna pattern directional ambiguity.

The 1965 NAS Woods Hole report (Reference 2) offered, among others, the following recommendations on long wave radio astronomy:

a. A space radio telescope with an aperture of the order of 20 km appears to be close in size to the ultimate for observations between a few MHz and a few hundred kHz. With larger apertures, effects due to irregularities in the interplanetary medium will, according to our best present knowledge, become the factor that limits the resolving power of the telescope.

b. Since the previous recommendation is for an ultimate space radio telescope, work should be started now that will lead to the use in space, within about 10 years, of a high-resolution broadband antenna system for radio-astronomical observations over the frequency range 10 MHz to a few hundred kHz.

The following guidelines were suggested (Reference 3):

a. Determine whether an antenna covering about 10 MHz to 500 kHz and having a beam area of about 100 deg^2 at 1 MHz is feasible and meets the scientific needs.

b. Consider alternative design concepts, among which should be the possible use of a simple or compound interferometer system to give added resolution within the main beam area.

c. Consider and compare locations for such an instrument, particularly as between a high orbit and a lunar base.

d. Consider this antenna as a possible payload for the Apollo Applications Program.

As we shall see in the next section, the crossed-H interferometer meets all these recommendations and suggestions.

During the second half of the study the above so-called "user requirements" essentially became design goals. It is pertinent to compare these design goals with the resultant capability of the crossed-H interferometer as it exists at the conclusion of the study. The data are shown on Table 3-1.

As can be seen, the requirements are all fulfilled with the possible exception of considering each end of the interferometer as a separate antenna for the strong time varying source made. The antenna pattern half power beamwidth vs. frequency for each end is as shown in Figure 3-1. However, as has been pointed out earlier, the solar and planetary sources are expected to be strong relative to the background; hence the feature that is of importance here is that the individual antenna pattern has a high front-to-back ratio, in order that any unwanted source interference can be excluded. The db ratio of each end is shown in Figure 3-1.

3.2 HYPOTHETICAL OBSERVATION PROGRAM. Described in this section is the hypothetical observation program used during the design of the antenna. It was formulated with the assistance of the consultant astronomers.

3.2.1 Typical Source Survey Mission. Sky survey missions are the most important task carried out by the interferometer. Resolution is the advantage over non-interferometer systems. Aperture synthesis analysis methods must be employed, utilizing computer synthesis of the radiation field. The resolution achieveable is close to the limit on resolution as theorized to be caused by interplanetary and interstellar scintillations, due to irregularities in the interplanetary medium causing multiple scattering. An empirical relation, evaluated by Erickson, gives the apparent angular size ϕ_o as 50 λ^2/r^2 in arc-minutes, where λ is the wavelength under observation in meters, and r the closest proximity pass distance of the ray from the sun in solar radii. This spreading of originally parallel rays from a distant source is subject to variations on the order of a factor of two due to solar cycle variations and interplanetary medium

Table 3-1. Crossed-H Interferometer Capability Compared To
Scientific User Requirements

	REQUIREMENT	CROSSED-H CAPABILITY
Life Time:	Minimum 1 year	2 or 3 years with resupply
Orbit Altitude:	Minimum synchronous	Synchronous circular
Effective Beamwidth:	10 deg^2 at 1 MHz — less than 10° in one direction; interferometer resolution to 2°	Interferometer 0.9° at 1 MHz for 10,000 m baseline
Pointing Accuracy:	1/2 beamwidth to 1/10. Pointing direction must be known to within 1/10 half power beamwidth.	±0.5° approximately
Pointing Stability:	1/10 beamwidth	±0.1°
Bandwidth:	500 kHz to 10 MHz	500 kHz to 10 MHz
Spectral Resolution:	Depends only on electronics	Fulfilled
Sensitivity:	Unfilled apertures	Fulfilled
Lock-on-Time:	1/2 second to several hours for time varying phenomena. For mapping observations, a drift-rate up to 1 deg/min.	Can lock on for several hours. Mapping drift rate 1° per 4 minutes.
Tolerance:	1/16 λ	Fulfilled
Front-to-Back Ratio:	Eliminate antenna pattern directional ambiguity.	Fulfilled
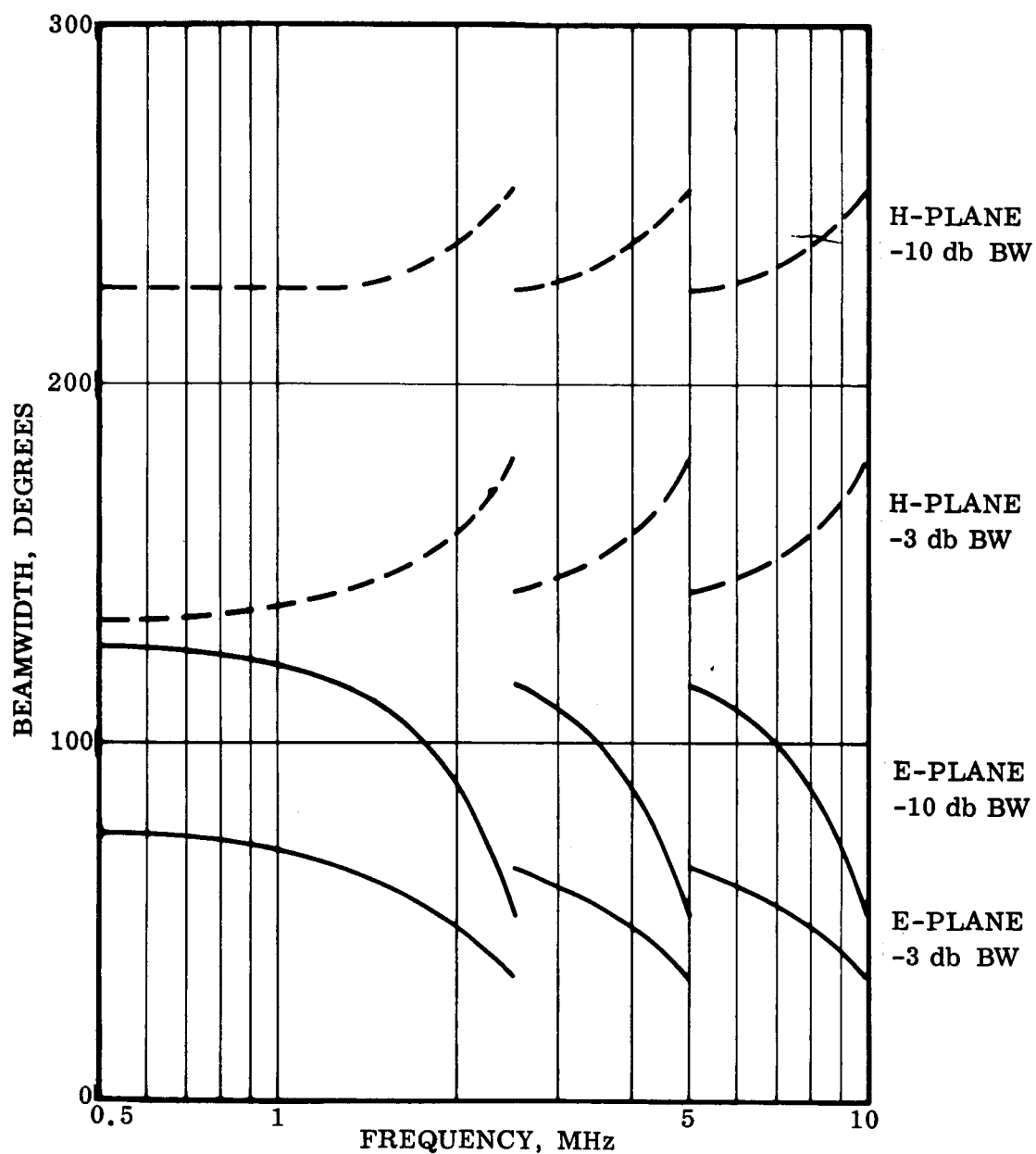
Figure 3-1. "Lazy H" on "Two-Element End-Fire Array" Beamwidth
For -3 and -10 db in Both Planes

anisotropy. This relation tells us that the lower the frequency, the larger the apparent size of the source. For the time period under consideration here this results in a maximum usable resolution of about $4°$ at 0.5 MHz, $1°$ at 1 MHz, and $1/4°$ at 2 MHz. Taking a possible error of a factor of two into account, we arrive at $2°$ usable resolution at 0.5 MHz where the crossed-H interferometer is capable of approximately $1.7°$, and $0.5°$ at 1 MHz, where it is capable of $0.9°$ with a 10,000 m tether.

The hypotehtical source survey mission is shown in Figure 3-2. It utilizes computer storage and analysis for aperture synthesis techniques. The reason for the three frequency ranges is, as explained elsewhere, that the boom and dipole length must be reset once for each frequency range to maintain coherent patterns throughout the entire 0.5 to 10.0 MHz band. The 5-10 MHz setting is called optional, since it may be very difficult to perform meaningful aperture synthesis with the radio frequency interference levels from the earth probable in this range. This interference could cause considerable "confusion," but flights made in the near future, such as the NASA Radio Astronomy Explorer, are expected to clarify whether or not aperture synthesis can profitably be performed in 5-10 MHz.

For aperture synthesis it is necessary to assume that all sources have constant emission during the mapping period. As envisioned (refer to Figure 3-2), a complete map of the sky at frequencies between 0.5 MHz and 10 MHz requires 428 days, including two orientations of the antenna pattern. The angular resolution obtained at the lower end of each frequency range is:

a.　$1.7° \times 1.7°$ (2.9 deg^2) or better normal to orbit plane.

b.　$1.7° \times 2.4°$ (4.1 deg^2) or better at $45°$ to orbit plane.

c.　$1.7° \times 14.1°$ (24 deg^2) or better in orbit plane.

Improved resolution, especially at the higher frequencies, may be achieved by using larger maximum tether lengths during the aperture synthesis.

3.2.2　<u>Strong Time Varying Source Observation Mission</u>. These types of observations can be carried out without the use of significant resolution, as long as the antenna employed has a considerable front-to-back gain ratio to avoid interference; e.g., it is necessary to discriminate between solar and Jupiter VLP radiation. Hence, the crossed-H interferometer arrangement can be used to advantage by considering each end as a separate antenna, each capable of receiving radiation in two polarization orientations. For circularly polarized sources, the pattern can be considered to look somewhat like a fat square pencil beam. The operations pertaining to this observational mode are depicted in Figure 3-3.

With the receiving systems as described in Section 4.2.1, and the antenna characteristics as shown in Section 4.4, the observational requirements for solar and planetary astronomy as depicted in Section 2.3.2 will be fulfilled. In operation, each end of the
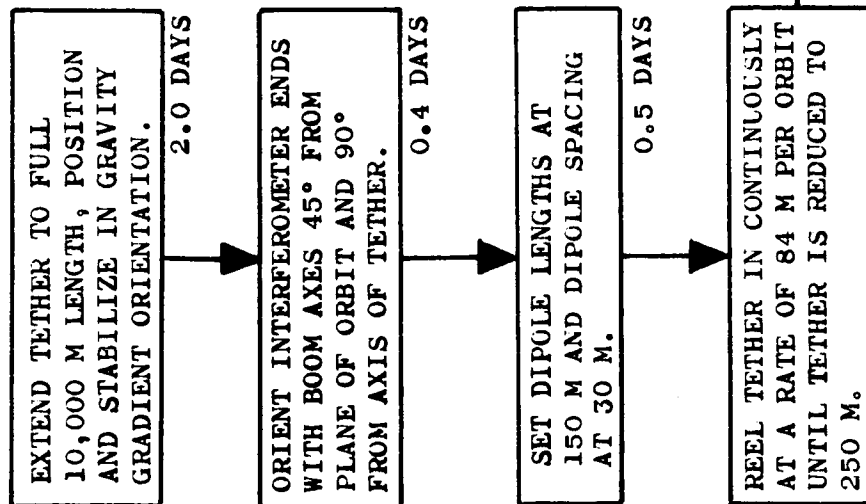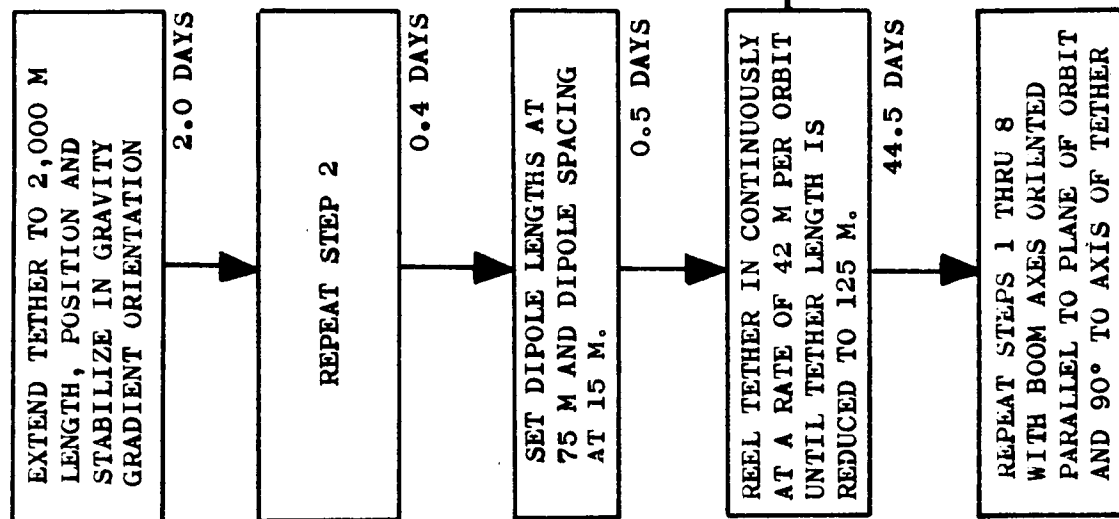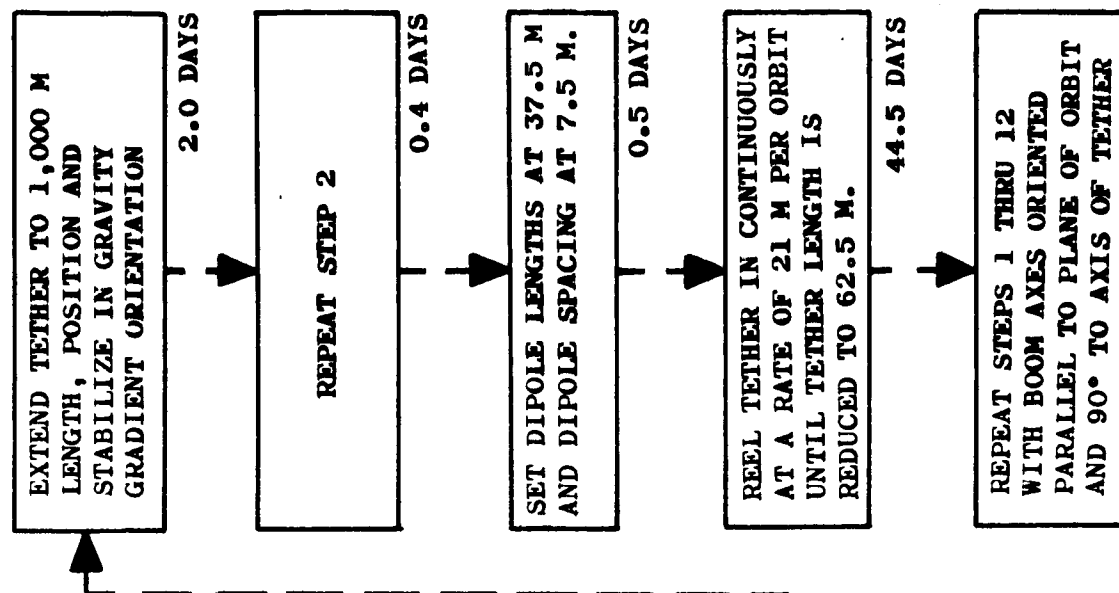
Figure 3-2. Typical Source Survey Mission

```
┌──────────────────────────────────────┐
│ WITH TETHER SET AT ANY CONVENIENT    │
│ LENGTH, ORIENT INTERFEROMETER ENDS   │
│ WITH BOOM AXES IN DIRECTION OF       │
│ TARGET SOURCE.                       │
└──────────────────────────────────────┘
                                240 MIN.
                    │
                    ▼
┌──────────────────────────────────────┐
│ SET DIPOLE LENGTHS AND DIPOLE SPACING│
│ OF ONE SATELLITE FOR THE 0.5 MHz TO  │
│ 2.5 MHz BAND AND THE DIPOLE LENGTHS  │
│ AND SPACING OF THE OTHER SATELLITE   │
│ SET FOR THE 2.5 MHz TO 5.0 MHz RANGE │
└──────────────────────────────────────┘
                                 20 MIN.
                    │
                    ▼
┌──────────────────────────────────────┐
│ CONTINUOUSLY ORIENT INTERFEROMETER   │
│ ENDS SO THAT BOOM AXES CONSTANTLY TRACK│
│ IN DIRECTION OF TARGET SOURCE FOR    │
│ PERIOD OF MEASUREMENT                │
└──────────────────────────────────────┘
                                  X MIN.
                    │
                    ▼
┌──────────────────────────────────────┐
│   REORIENT AND REPEAT FOR NEXT TARGET│
└──────────────────────────────────────┘
```

TOTAL TIME PER SOURCE X + 260 MIN.

Figure 3-3.   Strong Time Varying Source Observation Mission

array will be adjusted for one frequency range, so that the total range of 0.5-5 MHz or 2.5-10 MHz will be covered simultaneously. Thus, rapid, simultaneous data on all types of solar bursts and similar phenomena will be collected over a wide frequency range. This is a very strong factor in justifying the selection of the crossed-H interferometer arrangement.

3.2.3 Antenna Value vs. Orbital Life.

It has been mentioned that the crossed-H interferometer has been judiciously designed to be refurbishable and repairable through earth orbit rendezvous in order to extend its life time to several years. As Figure 3-2 shows, a source survey mission demands approximately 330 days (or 428 days for the entire frequency range of 0.5-10.0 MHz). It is certainly advantageous to then execute long period observations of strong time-varying sources, principally the sun, Jupiter, and perhaps also the earth. After the data from the first source mapping have been analyzed according to the aperture synthesis techniques, it would be advantageous to perform additional aperture synthesis investigations, this time with Jupiter or other interfering sources in different positions.

Increased astronomy value of the array with long lifetime would be in additional investigations undertaken after the orbit has precessed significantly and mapping another part of the celestial sky with the best resolution possible. However, a synchronous altitude orbit with an inclination of 28.5° precesses only about 4° per year. There remains the possibility to retract and dock the system, apply thrust, and bring it into an elliptical orbit. The new survey mapping would then cover a different part of the sky.

It is advantageous to launch the crossed-H interferometer so that its orbit plane will have a minimum inclination with respect to the ecliptic plane, in order that interference from the sun be minimized during the source survey modes.

SECTION 4

SYSTEMS DESIGN

4.1 CONFIGURATION

4.1.1 Launch Configuration and Deployment. In the launch configuration satellite dipoles and booms are fully retracted and the two center bodies are latched together at their respective intersatellite docking structures. The center bodies are thus spaced at 30 inches between faces, the dipole heads on each satellite space at 3 meters between dipoles and the dipoles completely retracted (Reference Figure 4-1).

This compact antenna assembly is mounted on the launch vehicle payload platform by eight pyrotechnic attachments located on the intersatellite docking structure. The payload platform in turn mounts on the four regular LEM attach points of a Saturn V vehicle.

In the launch configuration, the antenna assembly fits into a box of 1199 ft^3 or 29 percent of the available volume and weighs 4100 lb, or 17 percent of the available weight provided by the launch vehicle.

Deployment of the antenna assembly is illustrated in Figures 4-2 and 4-3. Views 3 and 4 show the CSM is docked to the payload platform, after the SLA has been retracted, and is used to withdraw the platform from the launch vehicle.

At this stage of deployment, the preliminary checkout is made whereby all possible tests of subsystem functions are accomplished. This timing permits the use of the CSM power and control systems, saving antenna power and propellant. It also maintains the antenna as an integral structure with the CSM, a great advantage if EVA for adjustment or replacement of equipment is shown necessary by these tests.

Upon satisfactory completion of this checkout, the antenna is oriented such that its booms are perpendicular to the local vertical. Then upon command from the CSM the antenna assembly is demated from the payload platform by activation of pyrotechnic attachments and springs. When it is free of the antenna the platform is removed by the CSM.

The CSM now flys around the antenna to inspect for damage from launch or deployment. Upon determination that all is in order, the astronauts proceed to final checkout.

The first step of the final checkout is to test the tether drive. The solenoid latch attachments between satellites are released and the satellites are separated by jet propulsion with the tether drive operating to permit the motion without improper tether fluctuations.
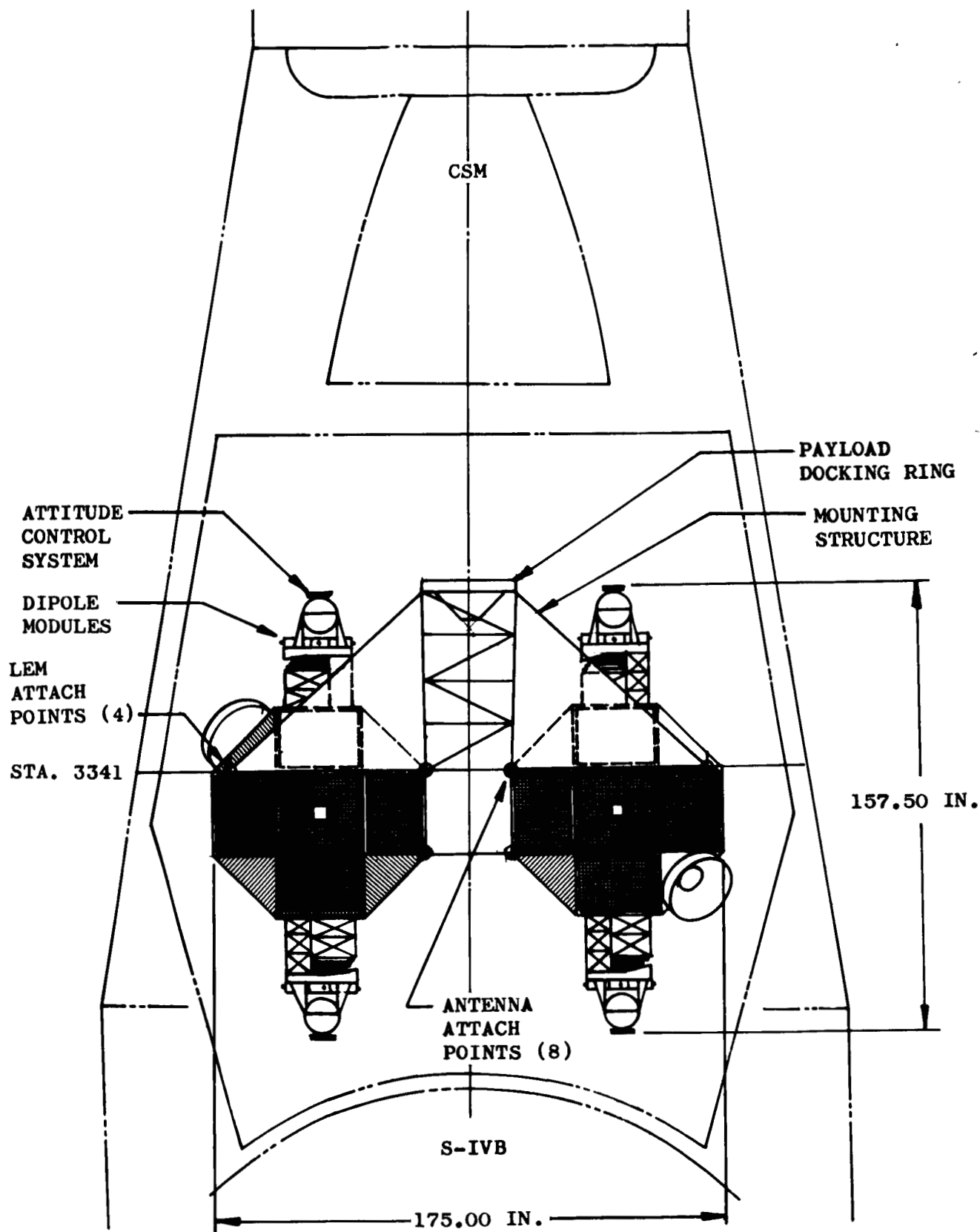
Figure 4-1.   Crossed-H Interferometer Launch Configuration (Sheet 1 of 2)
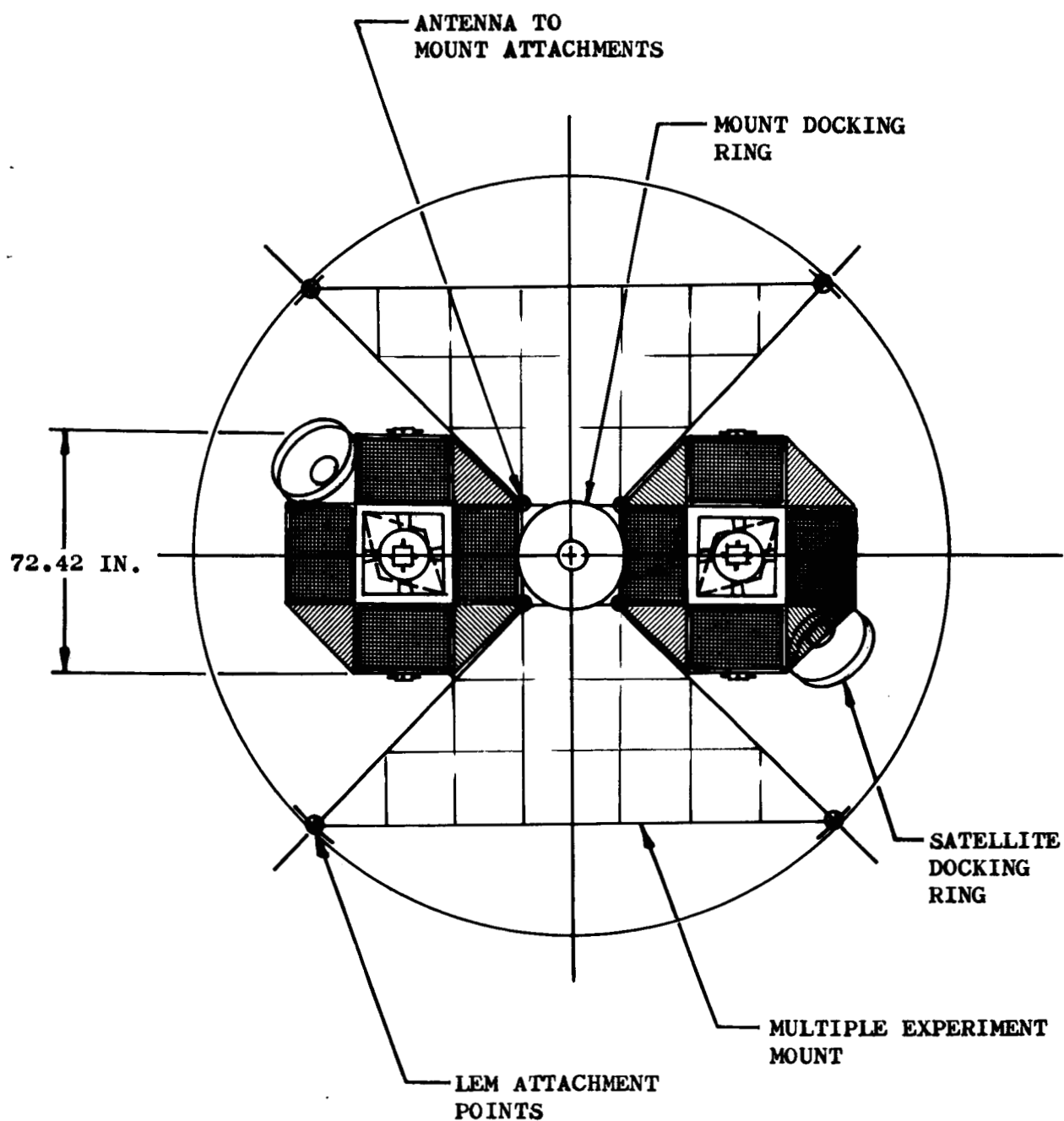
Figure 4-1.   Crossed-H Interferometer Launch Configuration (Sheet 2 of 2)
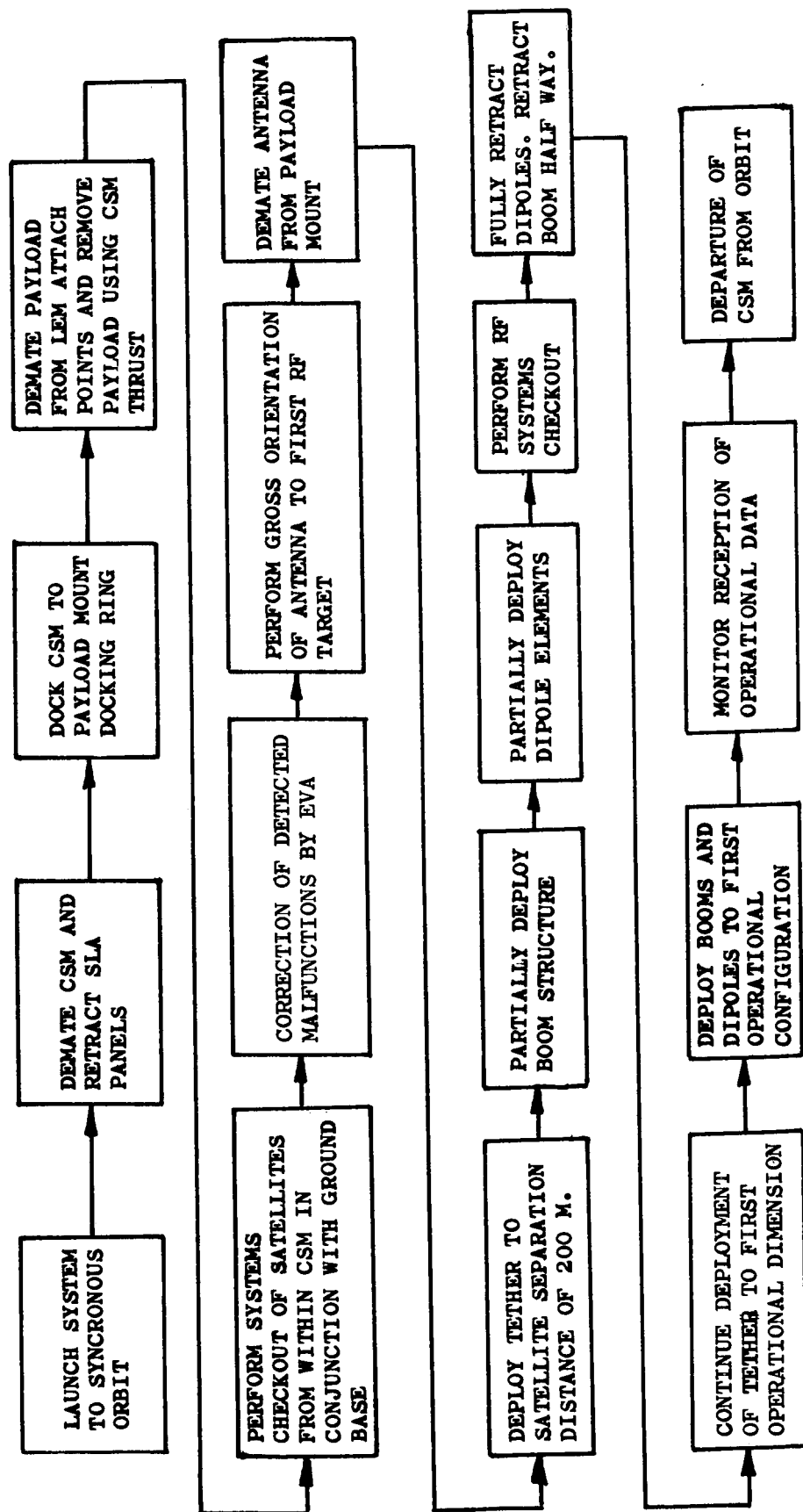
Figure 4-2   Deployment Sequence of Crossed-H Interferometer Radio Astronomy Antenna
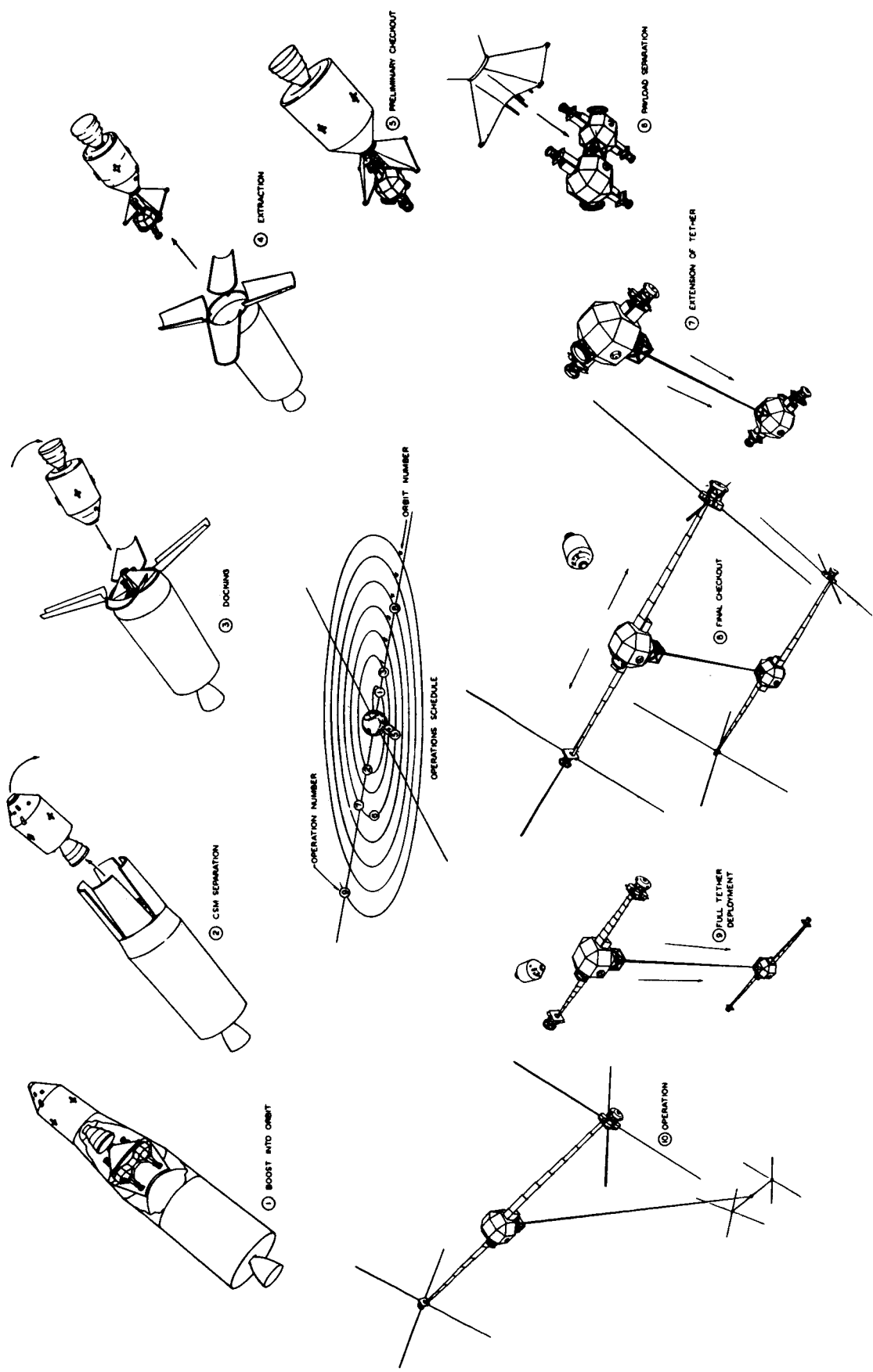
Figure 4-3. Cross-H Interferometer Orbital Deployment Operations

Extension is halted at approximately 200 meters between satellites and all subsystem checkouts are completed, excepting those dependent upon full tether extension. This extension length retains visual check capability and propinquity, should repair or replacement prove necessary.

The checkout is complete when the satellites have been extended the full 10,000 meters and the rf system has been tested over its full range. Operation is thereafter remotely controlled by ground station or is automatic. The CSM is free to depart for other missions.

4.1.2 Operational Configurations. The crossed-H antenna is deployed into a 28.5° synchronous orbit by a Saturn V launch vehicle system. Orbital operation of the antenna is charted in Figures 3-2 and 3-3. Briefly, the mission consists of mapping the celestial sphere in high frequency bands by invoking the technique of aperture synthesis by altering the length of the tether for each band. A second phase of the mission is the observation of specific strong time-varying sources in the two frequency bands simultaneously. An optional mode is the mapping of the celestial sphere by aperture synthesis in a third frequency band not as high as the other two.

In all modes the tether is oriented to the local vertical to utilize the environmental force of gravity gradient. This position is automatically sensed and is stabilized by jets.

In the first mapping mode, for the frequency band of 0.5 to 2.5 MHz, the dipoles and booms are fully extended to the 150m and 30m limits respectively. The booms are oriented to 45° from the orbital plane and 90° to the tether. The tether is started at the maximum 10,000m extension and steadily retracted at a rate of 8m per orbit down to 250m in 116 days.

For mapping in the second frequency band, 2.5 to 5.0 MHz, the dipoles and booms are set to 75m and 15m respectively. The tether is started at a 2000m extension and steadily retracted at a rate of 42m per orbit down to 125m in 45.5 days.

These two operations are then repeated with everything the same except the booms are oriented parallel to the plane of the orbit instead of at 45°. This attitude covers the remainder of the celestial sphere and provides complete coverage of the sphere in the two frequency bands in a total period of 333 days.

For the observation of strong time varying sources, the two booms are pointed at and lock on the source for the time required (unless limited by orbital factors). One satellite will be tuned to the 0.5 - 2.5 MHz range with dipoles and booms set at 150m and 30m respectively. The other satellite will be tuned to the 2.5 - 5.0 MHz range with dipoles and booms set at 75m and 15m respectively.

The optional mode for mapping will be tuned for the frequency band of 5.0 - 10.0 MHz with the dipoles and the booms at 37.5m and 7.5m respectively. The tether is started at 1000m and steadily retracted at the rate of 21m per orbit down to 62.5m to 44.5 days. By performing this operation twice, one with the booms parallel to the orbital plane and once with the booms at 45°, the celestial sphere can be mapped in this frequency range in 95 days.

The probability of completion of these operations is high for two primary reasons. First the design of the satellite utilizes proven design techniques and materials and freely incorporates redundant components and circuits.

Secondly, components liable to cause trouble, redundant or not, are designed to be replaceable by EVA. Such compensation for failure is provided for in four ways:

a. Redundant, wherein excessive or alternate components are include in the hook-up or can be included by automatic or remote control.

b. Receptive, wherein space and attachments are provided to permit and additional component to be added on, but requiring no disposal of the old unit.

c. Displaceable, wherein the old component is put aside to permit mounting an additional component but requiring no other disposal of the old unit.

d. Replaceable, wherein the old component is removed before the new one can be added and the old one must be disposed of.

Specific provisions for refurbishment of various components are indicated in Table 4-1.

If multiple failure or other malfunction should prevent satisfactory functioning of the antenna assembly the entire unit can be shut down and allowed to drift in orbit, utilizing the tether for station keeping. As soon as it is convenient to deliver an astronaut and parts, he can make repairs and replacements and restore the antenna to operating condition. A docking ring is provided on each satellite to facilitate such EVA.

One year in space is the design life of many components available in the immediate future, so that period has been established as the optimum for a general overhaul and refurbishment of the antenna.

To facilitate the activities of the astronaut, either satellite docks to the CSM and each can dock to the other. At major refurbishment the satellites are retracted to the 125m separation. The CSM then docks to one of them, the combination is stabilized and the other satellite is brought close. An astronaut exits with a remote control unit and, while watching from a safe vantage point, guides the satellite into dock.

The activities and effectiveness of the astronauts in the refurbishment operation are treated in some detail in Section 6 and Appendix I. However, even a cursory survey of the deployment, checkout, maintenance, and refurbishment functions of this

Table 4-1. Provisions for Refurbishment

COMPONENT	REDUN- DANT	RECEP- TIVE	DISPLACE- ABLE	REPLACE- ABLE
Tether Motor	X			X
Tether Drive	X			X
Tether Tape				X
Solar Cells	X	X		X
Batteries		X		X
Power Subsystem Electronics	X			X
Boom Drive Motor	X			X
Boom Drive Mechanism				X
Boom Drive Tapes				X
Boom Drive Gear and Spool				X
Lead-In Harness				X
Lead-In Tension Sensor	1			X
Lead-In Level-Wind and Motor	1			X
Dipole Motor				X
Dipole Drive				X
Dipole Assembly			X	X
ACS Nozzle and Solenoid Valve				X
ACS N/S and Regulator Assembly		X		X
ACS Propellant Tank		2		X

1. Redundant in that experiment can continue until astronaut can make repair.

2. Refillable.

mission clearly emphasizes the impact of man in space on the successfull completion of the mission.

4.2 STRUCTURAL/MECHANICAL DESIGN

4.2.1 Configuration Description and General Arrangement.   The antenna system consists of two variably tuned end-fire arrays, each mounted on a separate satellite with a variable length tether between.   The system provides high resolution over a wide frequency band through interferometer effect and dimensional variation.

Each satellite has its independent attitude control system, including horizon seeker and/or star trackers.   The propellant is cold gas expelled from 0.1-lb thrusters

located at boom tips and center body to give the six degrees of motion. Momentum wheels can be added when justified by mission requirements.

The tether between the two satellites is operationally extendible and retractable between 10,000 meters and 1000 meters. It serves two fundamental purposes; 1) it permits utilization of gravity gradient as a primary stabilizing force, thereby reducing propellant requirements, and 2) it provides a means of controlling and reducing the distance between satellites. For extensive EVA the two satellites can be docked together again into the launch configuration.

The center body encloses most of the mechanisms, including the tether drum and drive, and the electronics for the experiment and power system; solar cells are mounted on its outside panels. The fixed sections of the two booms are integrated with the center body structure.

These booms extend and retract by telescoping. They support the dipoles and operationally space them apart at 7.5 meters to 30 meters as required for the particular frequency under investigation. For maneuvering or EVA they can be fully retracted to the launch spacing of 3 meters.

The actual receiver elements of the antenna system are two dipoles crossing each other at the end of each boom of each satellite, thus forming an end-fire array at each satellite. Each half dipole can be controlled in extension or retraction operationally between 15 and 75 meters (30 to 150 meters overall). For maneuver or EVA they are fully retracted for protection.

The above configuration is the result of many tradeoffs, a few of which may be reconsidered upon further analysis. Attitude control and electronic tradeoffs are treated in their respective sections.

The reasons for adopting the tether are noted above and the choice of tether form is explained in Section 4.1.2.1.

The symmetrical shape of the center body was chosen to approximate a balance in solar exposure and the flat panel modification from a sphere was chosen as it provides an exposure that is as effective or better, and it permits a simple modular construction of solar cell panels and structure.

The silicon solar cells with glass cover are chosen over the cadmium/sulphide cells. It appears the 13.0 watts per square foot anticipated by 1970 greatly surpasses the lighter cadmium/sulphide cells, and space is more critical than weight. The panels themselves are built of corrugation and zees as this method of construction dissipates the heat more effectively than honeycomb.

Three types of boom construction were considered; extendible mesh tube, square section truss, and triangle section truss. The extendible tube type was discarded because of poor torsional rigidity and the space requirement for spools and section transformation. The triangular truss was chosen over the square as it is lighter and has fewer pieces, is more rigid in torsion and, with proper design of webs, is less subject to thermal distortion. The webs are designed for maximum pass-through of solar energy to obtain a minimum temperature gradient and resultant distortion.

The boom drive was the result of many considerations. The telescoping characteristic dictated an external drive, as space for and access to an internal drive is severely compromised. Specific positioning for each boom section is desirable to prevent boom sections from floating at intermediate positions with resultant unbalance and possible inertial problems.

Tape was chosen over cable as it is more flexible and permits a small single-width drum. The drum was chosen over the pulley as it permits the introduction of travel ratio between sections as discussed below. The gear drive has large enough pitch and sufficient excess strength to meet reliability standards.

For the antenna dipoles there appears no alternate for the extendible tube concept. However, to avoid excessive distortion due to temperature gradient it is necessary to use a wire mesh; it permits penetration of solar energy to the far side and permits selective application of materials for the lateral and longitudinal wires, as discussed in Section 4.1.2.4.

4.2.2 Detail Design

4.2.2.1 Inter-satellite Tether and Controls.
The tether between the two satellites represents a special type of structure. It is subject to the tensile load due to the gravity gradient differential force between the satellites, certain dynamic transients and a possible side load due to solar pressure.

The basic load is in the order of 0.02 lb. The dynamic transients, although indeterminate at this time, are assumed to be no more than several pounds (see Section 4.4.2).

A flat mylar tape is used in lieu of wire to ensure structural integrity if it is struck by a meteoroid. Solar pressure on the flat tape is minimized by orienting its edge to the sun. One half of each wide face is colored black and the other half silver. The increased solar pressure on the silver half over that on the black half creates a torsion that rotates the edge of the black half into the sun. In this attitude the maximum solar pressure on the tape is approximately 0.49 lb.

The resultant lateral deflection is a function of various dynamic effects including tape tension. These deflections have been determined to be negligible, as confirmed by the reference noted in Section 4.4.2.

An added advantage of tape is the ability to store the tape on a single-width reel. This method eliminates the volume loss due to gaps between wraps. It also eliminates the need for a level winding device and its complexities.

The tape selected for this application is 0.0015×1.0 in. Mylar G with an ultimate tensile strength of 40,000 psi, density of 1.395 gm/cm^3, and modulus of elasticity of 800,000 psi.

The 10,000m of tether tape is stored equally divided between the two satellites on identical reel systems. The tether runs into the satellite to a guide roller than normally directs the line of tether force through the satellite center of gravity. From that roller the tether runs through a tension sensing device and then into the drum, which is mounted with its axis on the satellite vertical centerline to avoid unbalance with change in tether wrap (see Figure 4-4).

The center body structure is sufficiently open to permit the tether to deviate through a cone of 20° without touching the adjacent structure during pitch and roll maneuvering of the satellite. The guide rolls are placed as close as possible to the center of gravity of the satellite, thus minimizing the displacement of the line of tether force and the resultant torque on the satellite during maneuvering deviations.

The tension device consists of a guide roller mounted on a pivoted spring loaded arm. The resultant force of the tape tension acting on the guide roller rotates the arm against the spring. At the travel limits electrical switches command the reel motor to make the necessary revolutions. A rotary potentiometer at the arm pivot provides a modulated tether tension readout.

To improve reliability as well as to facilitate refurbishment, each reel has the capacity to hold a little more than the entire 10,000 meters of tether. Normally each reel winds up not more than 5000 meters but if one reel drive system should fail when empty, the other reel and drive can continue the full function.

At the time of major refurbishment the tether may be replaced should further study reflect the desire to do so. The replacement procedure would be as follows:

a. The astronaut firmly attaches a transfer reel onto the docking structure between the two docked satellites. This reel contains the replacement tether and an empty spool to receive the old tether.

b. The old tether is completely reeled onto one of the satellite reels if not already in that condition. The astronaut disconnects the prepared splice near the empty reel and connects the prepared splice of the replacement tether. He then connects the loose end of the old tether to the empty spool of the transfer reel.

c. The empty satellite reel is then rotated and filled from the transfer reel. The old tether is wound onto the replacement unit reel at the same time.

1. STORAGE DRUM
2. DRIVE MOTOR
3. SPEED REDUCER
4. TENSION SENSOR
5. PIVOT ROLLER
6. BI – COLORED TETHER

MAXIMUM TETHER
DISPLACEMENT ANGLE

20°
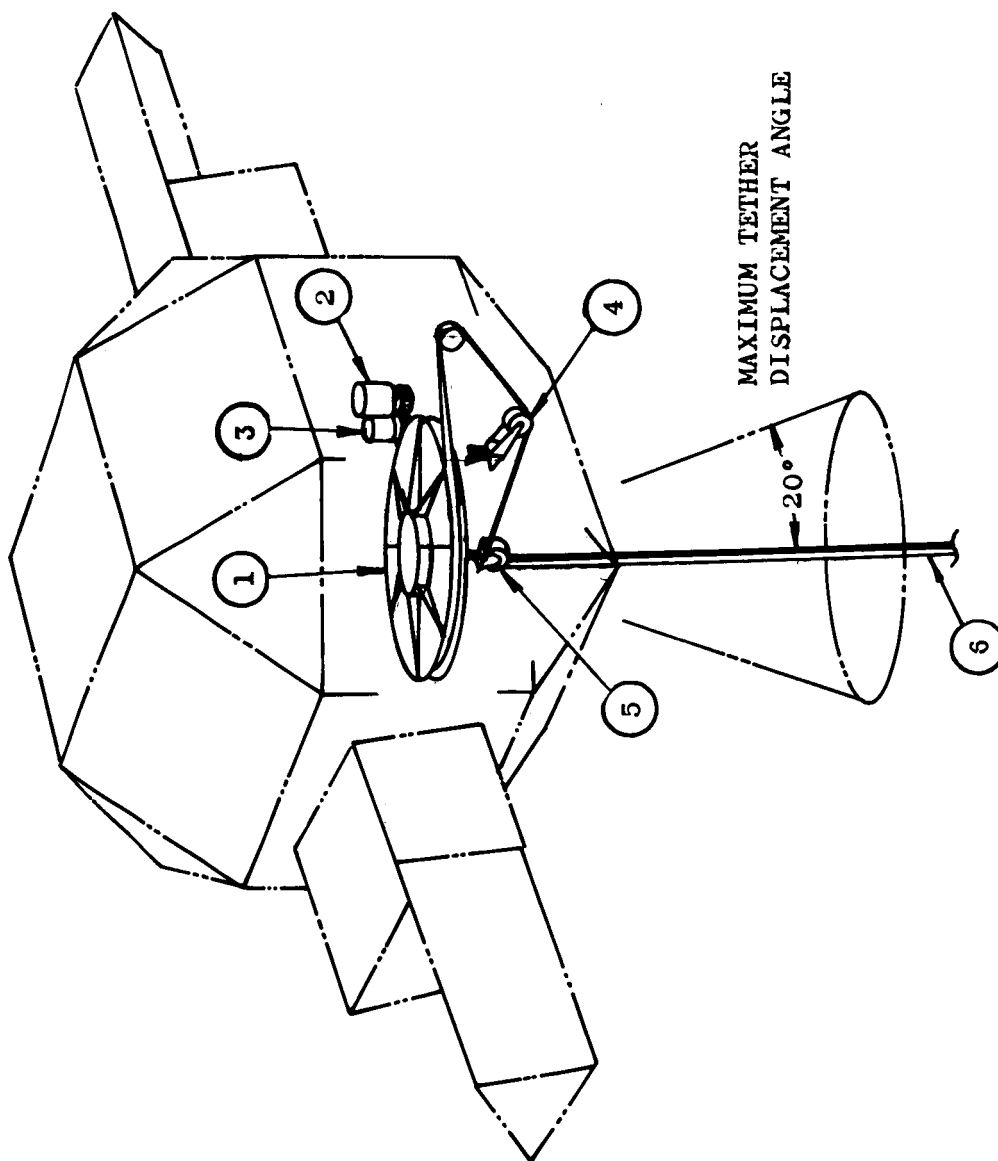
Figure 4-4. Tether Control System

d. When the old and new tethers are fully transferred the astronaut disconnects the old tether at the now empty satellite reel and connects the new tether.

e. The transfer reel is returned to the SM.

4.2.2.2 Center Body. The center body itself is a typical lightweight structure. It is octagonal in section and is comprised of a central webbed beam of parallelogram section and two major bulkheads of plate/stiffener construction. All material is aluminum alloy (see Figure 4-5).

The central beam is two adjacent booms of triangular section that serve as the fixed portions of the telescoping booms and are of the same cap and web construction as the telescoping sections.

Each exterior panel of the center body is faced with solar cells. To obtain maximum power supply for the area available the cells are silicon cells 0.04 in. thick with 0.001 integral glass cover which has a capability of 13.0 w/ft^2. The total average area of 115 ft^2 furnishes an average power output of 330 watts per satellite.

The cells are mounted on a substrate that in turn is mounted on a corrugated aluminum panel, which is reinforced with stiffeners and edged with channel members. Solar cell panels open on hinges to provide EVA access to replaceable equipment within the center body. The smaller triangular panels are fixed.

The solar cells are refurbished by placing add-on panels over the original ones with quick fasteners that do not prevent subsequent opening of the panels. They also contain built-in electrical connectors that engage as the panel is put in place.

Hand and foot rails are provided at the edges of the square panels as necessary to provide the astronaut a firm attachment from which he may accomplish his various EVA tasks (see Figure 4-6).

To the inside face of these hinged panels, where it can be highly exposed for access, much of the electronic equipment is mounted. The individual pieces are assembled on modular trays that are attached to the exterior panel reinforcing zees by quick fasteners. Maintenance and refurbishment is accomplished by replacement of the modules as required.

To ensure proper heat transfer and passive thermal control the modular trays are limited to about four 10-in. squares on each 30-in. square panel.

Sufficient depth of zee, spacing of modules to equalize "see-thru" areas, and thermally insulated attachments provide an internal ambient temperature of approximately 150° F in direct sunlight.
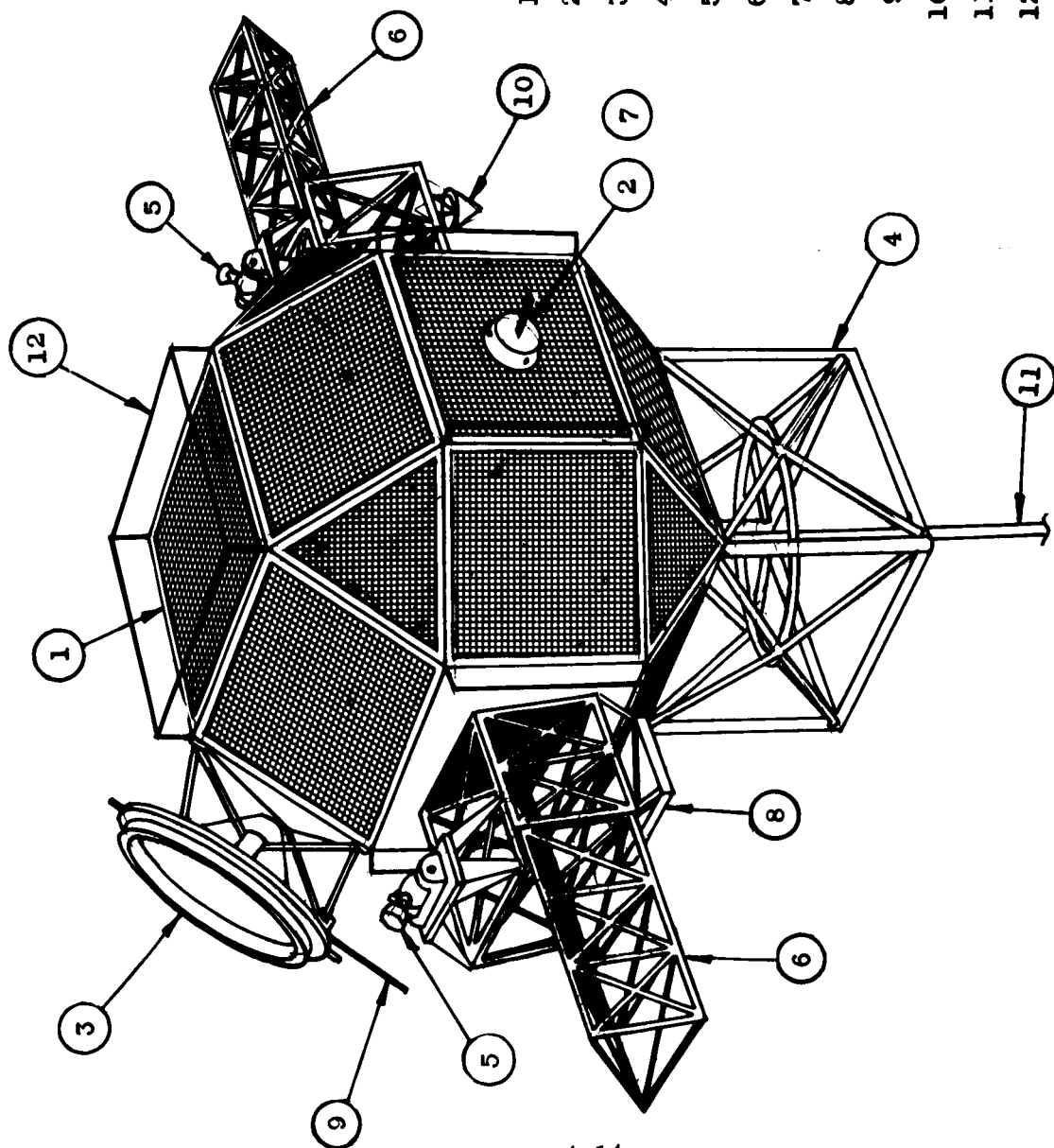
1. SOLAR CELL PANELS (22)
2. ATTITUDE CONTROL MODULES (2)
3. CSM DOCKING RING
4. INTER-SATELLITE DOCKING STRUCTURE
5. STAR TRACKERS (2)
6. BOOM STRUCTURE
7. TELEMETRY RECEIVING ANTENNA (2)
8. RANGE, D/F AND RELAY ANTENNA
9. CSM DOCKING ANTENNA
10. TLM TRANSMITTING ANTENNA
11. TETHER
12. EVA HAND AND FOOT RESTRAINT RAILS.

Figure 4-5. Center Body General Arrangement

4-14
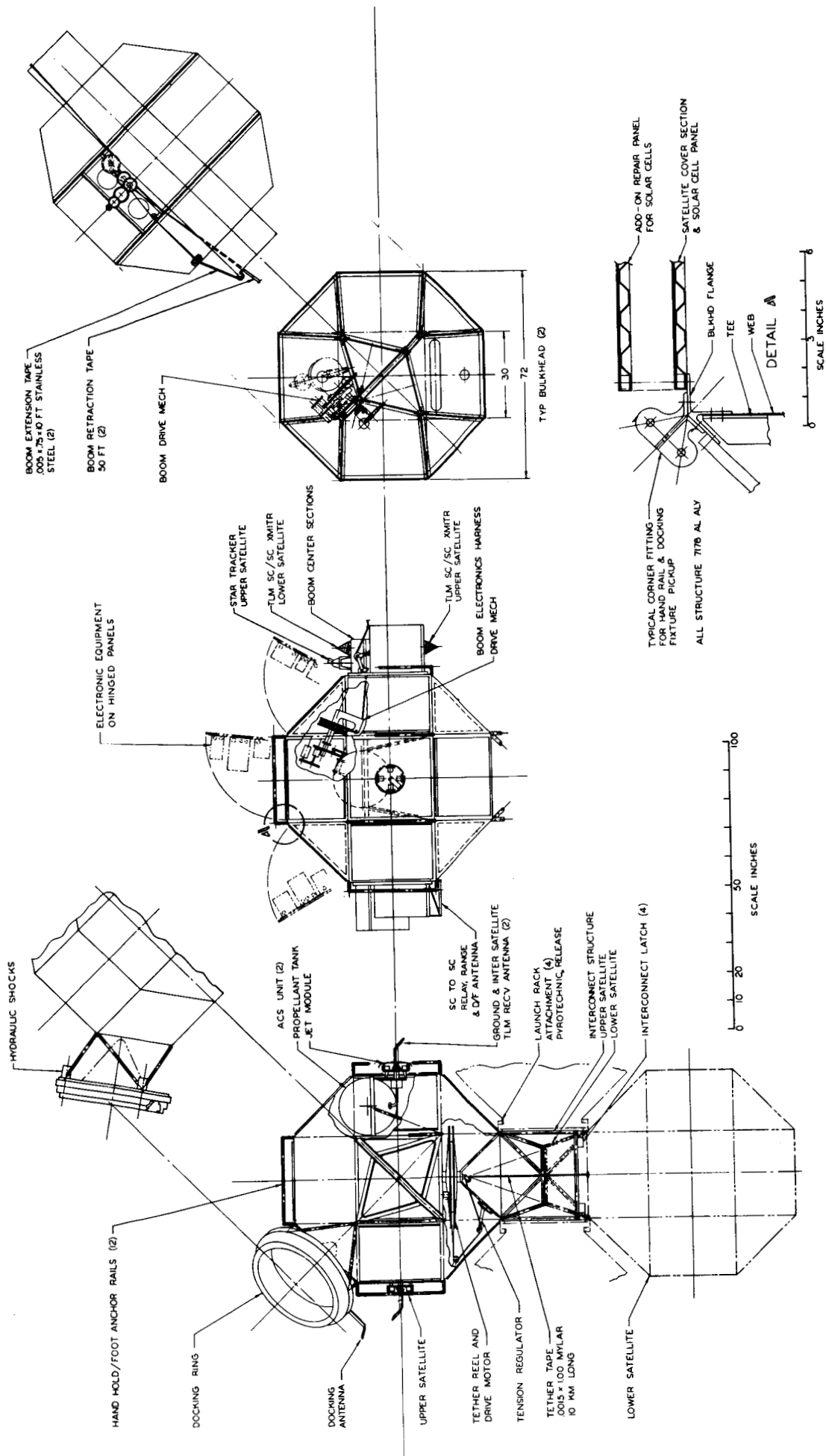
Figure 4-6. Center Body Installation

BOOM EXTENSION TAPE
.005 × 1.75 × 10 FT STAINLESS
STEEL (2)

BOOM RETRACTION TAPE
50 FT (2)

BOOM DRIVE MECH

STAR TRACKER
UPPER SATELLITE

TLM SC/SC XMTR
LOWER SATELLITE

BOOM CENTER SECTIONS

TLM SC/SC XMTR
UPPER SATELLITE

BOOM ELECTRONICS HARNESS

BOOM
DRIVE MECH

TYP BULKHEAD (2)

30

72

ADD-ON REPAIR PANEL
FOR SOLAR CELLS

SATELLITE COVER SECTION
& SOLAR CELL PANEL

BLKHD FLANGE

TEE

WEB

TYPICAL CORNER FITTING
FOR HAND RAIL & DOCKING
FIXTURE PICKUP

ALL STRUCTURE 7178 AL ALY

DETAIL A

SCALE INCHES
0   3   6

ELECTRONIC EQUIPMENT
ON HINGED PANELS

ACS UNIT (2)

PROPELLANT TANK
JET MODULE

SC TO SC
RELAY, RANGE
& DVF ANTENNA

GROUND & INTER SATELLITE
TLM RECV ANTENNA (2)

LAUNCH RACK
ATTACHMENT (4)
PYROTECHNIC RELEASE

INTERCONNECT STRUCTURE
UPPER SATELLITE
LOWER SATELLITE

INTERCONNECT LATCH (4)

SCALE INCHES
0   10   20   50   100

HYDRAULIC SHOCKS

HAND HOLD/FOOT ANCHOR RAILS (2)

DOCKING RING

DOCKING
ANTENNA

UPPER SATELLITE

TETHER REEL AND
DRIVE MOTOR

TENSION REGULATOR

TETHER TAPE
.0015 × 1.00 MYLAR
10 KM LONG

LOWER SATELLITE

A docking ring is provided on each center body to facilitate EVA. With the CSM firmly docked the satellite becomes integral structure which reduces hardship and hazard for the astronaut. Hydraulic shock absorbers are provided to reduce the acceleration of the satellite when contacted by the CSM.

Assuming a CSM velocity of 1 ft/sec, an allowable satellite acceleration of 0.1 g at a single satellite weight of 2500 lb, the travel of the shock absorbers is 1.9 in. To allow for the possible case where docking is required for the two satellites combined, this stroke must be increased to 3.8 in.

Once the docking is complete the shock absorbers should be locked out to provide positive connection for attitude control purposes. This is done by a manually operated cam device operated by the astronaut when he just emerges from the CM. It could also be done electrically with screw jacks, but this would probably reduce reliability.

Provision for docking the two satellites together is mounted on the appropriate panel of each satellite. One satellite carries a probe and the other a tapered seat, both constructed of welded aluminum alloy tubing and automatically locked together by a solenoid latch.

To dock the satellites the astronaut positions himself in a safe vantage point on the satellite attached to the CSM and by remote control unit guides the free satellite into the dock. With the astronaut in EVA he is so intimately in control of the relative velocity of the docking satellite simple bumper pads are considered all that is necessary for shock absorption.

At the base of this docking structure there is provision for pyrotechnic bolts for attaching the satellites (docked) to the launch vehicle payload rack. Once these four attachments are fired, as discussed in Section 4.1.1, they serve no further function.

A jet cluster is mounted on each of two sides of the center body so positioned that it can provide rotation about the centerline of the boom or the tether, or provide translation along the same centerlines. This jet cluster uses the same units as those at the boom ends but are arranged to be mounted through a solar cell panel.

The jet/solenoid valve unit can be replaced individually through EVA by unclamping a toggle and unplugging propellant and electrical quick disconnects. At time of refurbishment the entire cluster, including four jets and their regulator, can be replaced by adding on a new module containing the replacement components. The astronaut then inserts four attachment pins and two disconnects, one for power and one for propellant.

The attitude sensors are all mounted on the center body and fixed sections of the booms. They consist of star trackers, horizon sensor, inter-satellite range and direction finders, as well as dipole array position sensors.

The upper satellite has two star trackers mounted on the top of the boom sections oriented at 90° to each other and in a direction that will permit one tracker to sight a star near the polar axis. The lower satellite has one star tracker that can sight a star near the polar axis and one horizon sensor to relate the satellite to earth.

The inter-satellite range and D/F receiver and transmitter antennas, to relate one satellite to the other, are mounted on the fixed boom of each satellite such that the antennas face each other.

Other antennas mounted on the center body are the docking antenna, consisting of a 6 in. blade located at the docking ring, and the telemetry sending and receiving antennas. The sending antenna is a 6-in. conical helix located on the center body where it can see the jet assemblies on the other satellite center body where the corresponding receiving antennas are located.

These telemetry receiving antennas project from the center of the jet assemblies in a manner such that they function with or without the jet add-on modules. Like all the protruding antennas these are flexible and rubber coated to prevent damage to the astronaut or his suit.

The boom drive mechanism is located within the center body. It consists of spools for the tapes that drive the booms, redundant drive motors, and a 2-way dogged clutch.

The boom extend tapes are wound on separate spools, one for each boom, on a common torque shaft. This shaft carries a wide gear that engages the toothed output of the clutch plate. The clutch plate has dogs on each face that engage with the output of one (or the other) of the drive motors. Engagement with one and disengagement with the other is accomplished by longitudinally moving the spindle to which the clutch plate is attached, causing one set of dogs to engage while the toothed output slides along the teeth of the wide gear on the spool shaft.

The spindle motion causing the selective clutch engagement is actuated by an over-center sping-loaded lever. The spring holds the clutch in engagement with the first motor as long as it functions properly. However, on command from a control station a solenoid is energized and it pulls the lever back over dead center. This motion slides the spindle, disengages the first motor and engages the second motor. With the lever having passed over center the same spring holds the clutch in this engagement when the solenoid is de-energized.

The dog clutch is selected as being immune to space environment with the possible exception of lubrication on the sliding spindle and the one-shot solenoid. These in turn can be selected to give a high degree of reliability.

The boom retract tapes are wound on separate spools mounted on a common shaft driven by the extend shaft through two gear pairs. These gears provide approximately an 8/1 increase as the retraction tape is attached to the dipole head and therefore moves approximately eight times as fast as the extend tape.

However the ratio of revolutions of the extend to the retract spools is not constant due to the change in effective spool diameter with take up or pay out. To compensate for this variation and yet keep the right tension on the tapes the retract spools mounted on their common shaft are rotated by the shaft only through negator springs which will control the tension.

As the retract tape is tied to the dipole head its motion is the same. Consequently this tape is passed through a digital counter which reports the exact boom extension and provides a means of actuating automatic stops.

The electrical harness reels are also located within the center body. The harness itself is approximately 3/8 in. diameter and covered with a flexible tube of teflon or equivalent. It contains two coaxial cables for the rf signals and all other electrical leads insulated as necessary.

The harness runs in from the dipole head with supporting stand-off rings at each boom section end and passes over a tension regulator. The regulator pulley consists of a pulley mounted on a pivoted and spring-loaded arm such that the spring imparts a near constant tension on the harness. As the spring load approaches predetermined limits electrical contacts are completed commanded in the reel motor to take up or pay out the harness.

The harness reel is fixed and the motor drives a rotating winding guide with a translation introduced by a cam to level the wraps. The rotating guide has a slip clutch between it and the motor; in case the motor or speed reducer jams the harness can be pulled off the reel by boom extension. The clutch consists of spring-loaded balls applying torque through detents in the clutch plate.

Once the harness has been pulled out and the drive fails, the retracting boom will cause slack loops in the cable that are undesirable for normal operations but they do not prevent antenna functions. As soon as convenient the astronaut can replace the faulty unit and the tension will again be properly controlled.

4.2.2.3 <u>Boom.</u> The boom that supports the crossed dipoles is a typical space structure in that it extends for considerable distance, is lightly loaded, and must be essentially impervious to the space environment of vacuum and temperature extremes. Moreover, in accord with the basic concept of this antenna, it must be extendible and retractable to permit tuning the antenna for a wide frequency range, and therefore it incorporates moving joints and mechanisms that must have a high reliability over an extended period.

The basic structure is a triangular section boom with three cap strips and three webs. The webs are perhaps the most characteristic feature for they consist of little more than diagonal tension strips minimized in area to permit the maximum exposure to the sun at the far side of the boom.

The web compression members are not positioned parallel to each other but are individually slanted such that at no time can the shadow of one completely cover another and thereby create a large ΔT. This configuration, together with selective use of white paint on the outside and absorptive paint on the inside, reduces the distortion due to thermal gradient to a level compatible with the accurate pointing requirements of this antenna.

To provide the capability of extension and retraction, the boom is divided into eight sections, seven of which telescope into the eighth that is part of the fixed structure of the center body. To avoid the friction from members sliding under bending and shear loads, each telescoping section rides on two or more ball-bearing rollers at each of its three caps (see Figure 4-7).

Torsion is transmitted as a side load on these rollers as the magnitude is sufficiently small to be acceptable. However, in view of this sliding action and the possibility of a roller jamming, the sides and bottom of the cap groove in which the rollers operate are coated with a dry lubricant.

Two of the rollers at any given station are fixed but the third is adjustable to permit optimum roller clearance for freedom of motion and elimination of mechanical play. It is anticipated that this adjustment will not need alteration in orbit, but if desired, it may be used as a test of the astronaut's ability to make vernier adjustments with a small wrench.

The boom drive consists of a tape driven gear and rack. The tape is 0.005 × 0.75 in. stainless steel attached by pip pin to the adjacent inboard boom section and wound on a drum mounted integrally with the gear by a pip pin axle pin. Thus tape drum and gear can be readily replaced at refurbishment, all being identical except for the inboard tape length, but is is not anticipated such replacement will be necessary.

The extreme inboard section of the drive tape is wound on the boom extension drum as discussed in Section 4.2.2.2 (see Figure 4-8).

4-19
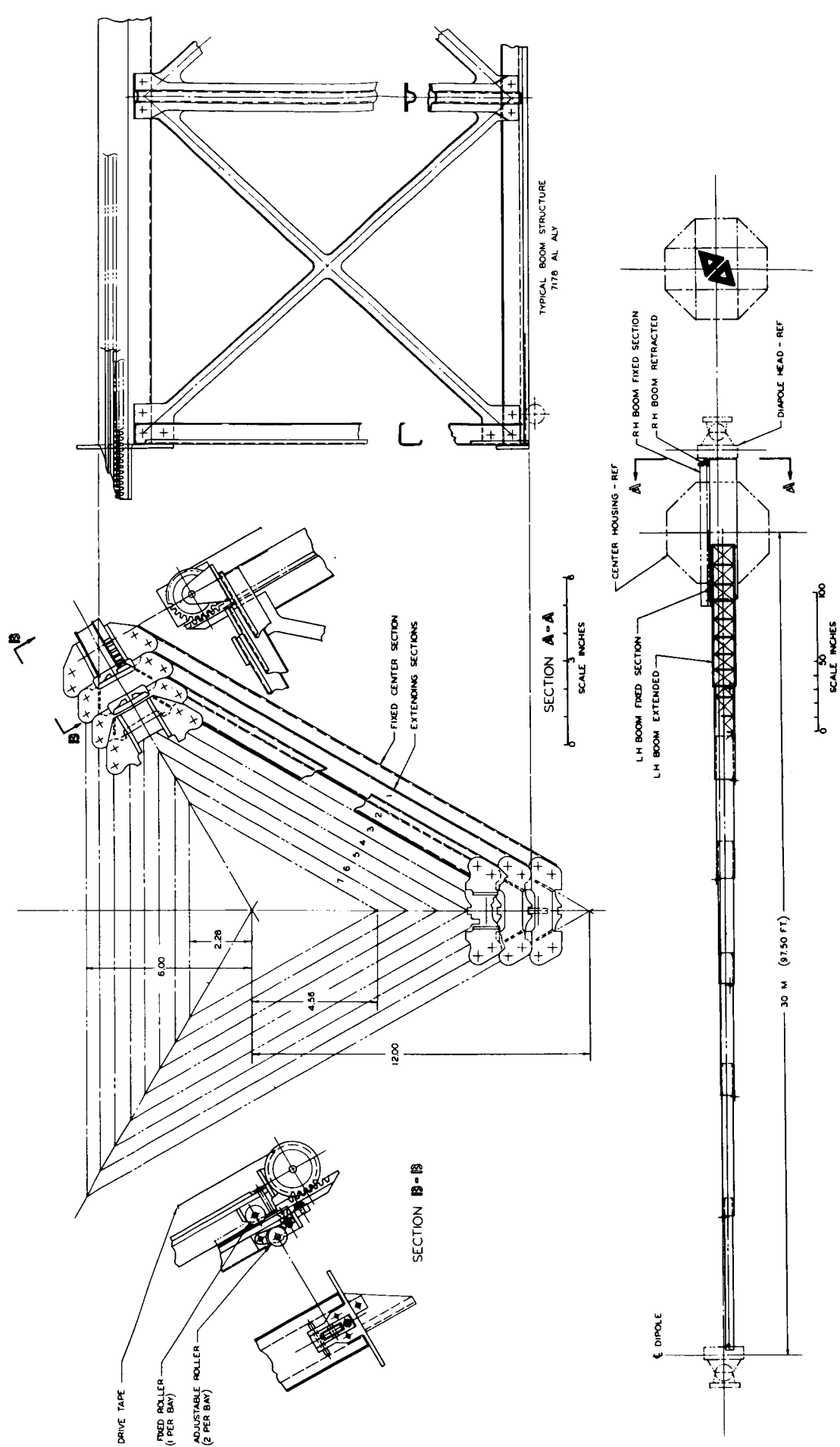
Figure 4-7. Boom Installation
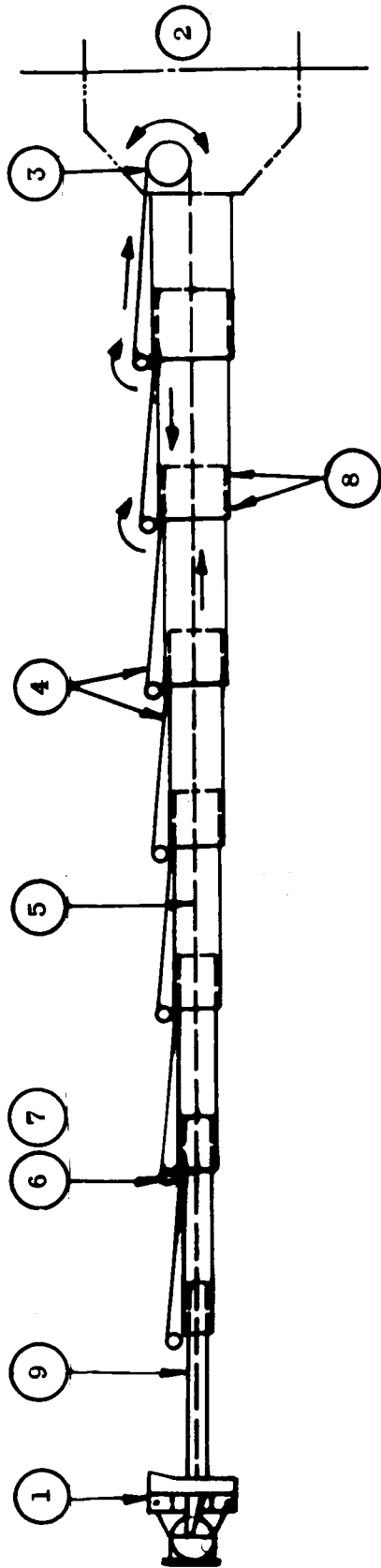
1. DIPOLE HEAD
2. MAIN BODY
3. MOTOR AND CLUTCH SYSTEM
4. EXTENSION TAPES
5. RETRACTION TAPE
6. DRIVE GEAR
7. TAPE DRUM
8. INTERNAL ADJUSTABLE ROLLERS
9. GEAR RACK
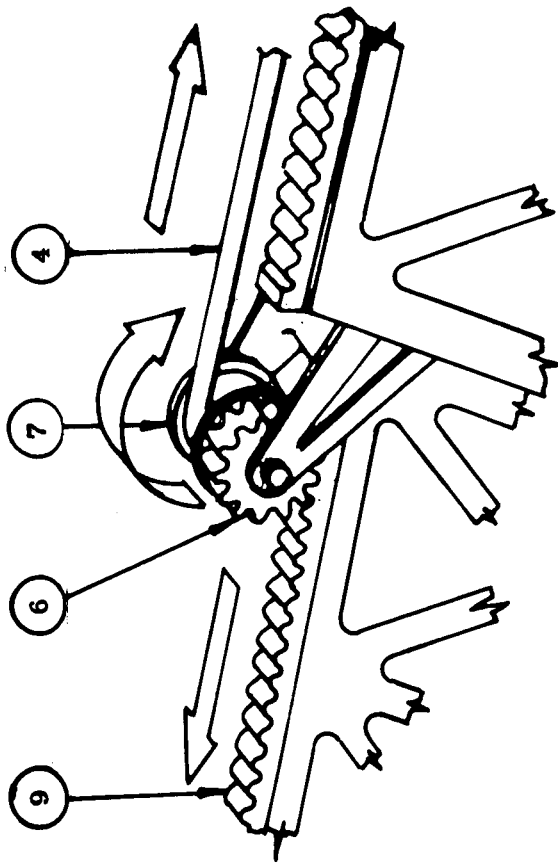
Figure 4-8. Boom Actuation System

It should be noted that the drum-gear combination permits the introduction of a relative travel ratio between adjacent boom sections of 1.06. This is significant because all boom sections must be stored for launch in the fixed section and therefore are essentially the same length. However the outer sections do not require the overlap necessary for the inner sections and should therefore travel further for optimum structural efficiency.

All structural and mechanical parts are aluminum alloy excepting the tape, ball bearings, rollers, and attaching parts. The teeth of both gear and rack are coated with dry lubricant Lubeco or equivalent.

Various types of extendible boom structures were evaluated prior to the selection of the triangular truss as the optimum design configuration.

Figure 4-8a illustrates four of the alternate structural concepts studied.

The first concept evaluated has an extendible tubular mesh boom. In this case the structural concept proved to be extremely light but exhibited very poor structural properties in torsional stiffness, thermal distortion also proved to be in excess of the desired $5°$ or less of total boom deflection, due to the poor heat transfer from the exposed side to the shaded side for the large tube diameter. The telescoping tube structure was excellent but it was excessively heavy and very poor in thermal distortion due to the heavy gage wall thickness, which did not adequately transfer temperature gradients from the exposed to shaded sides of the tube sections. The thermal deflection in this case was so severe that roller binding would result unless looser tolerances were used, which would add undesirable mechanical deflection to the structure.

The square open truss section proved to be acceptable both structurally and thermally but proved to be heavier than the final concept, the triangular open truss.

An added drawback to the tubular and square open truss is the fact that their center sections can not be mounted back-to-back within the center body without greatly enlarging the dimensions of the center body.

4.2.2.4 Dipole Head, Dipoles, and Jets. The dipole head structure mounted on the outer end of the boom consists essentially of two machined flanged plates and trussed webs, all of aluminum alloy. It supports the dipole assemblies and electronic assemblies, and mounts four stand-off brackets that support the propellant tank and jet assembly structure (see Figure 4-9).

The four dipole assemblies are identical and are driven by the common motor and central drive gear system. The dipole assembly module contains the dipole stored on a spool and the mechanism necessary to extend and retract the dipole on command. It also contains a digital length counter that reports to the command station the actual dipole length extended.

STIFFNESS	TORSION	THERMO DISTORTION	WEIGHT
POOR	POOR	POOR GOOD	VERY LIGHT
GOOD	GOOD	POOR	HEAVY
GOOD	GOOD	FAIR	LIGHT
GOOD	GOOD	GOOD	LIGHT

500/1 10/1 10/1 10/1
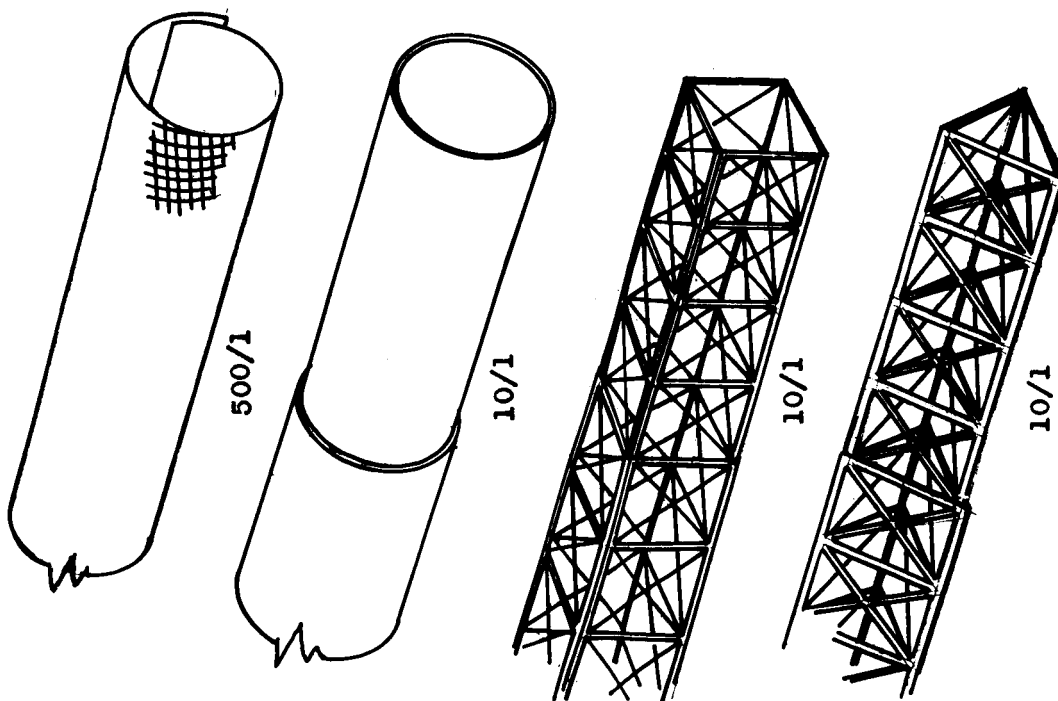
Figure 4-8a. Boom Structure Comparisons
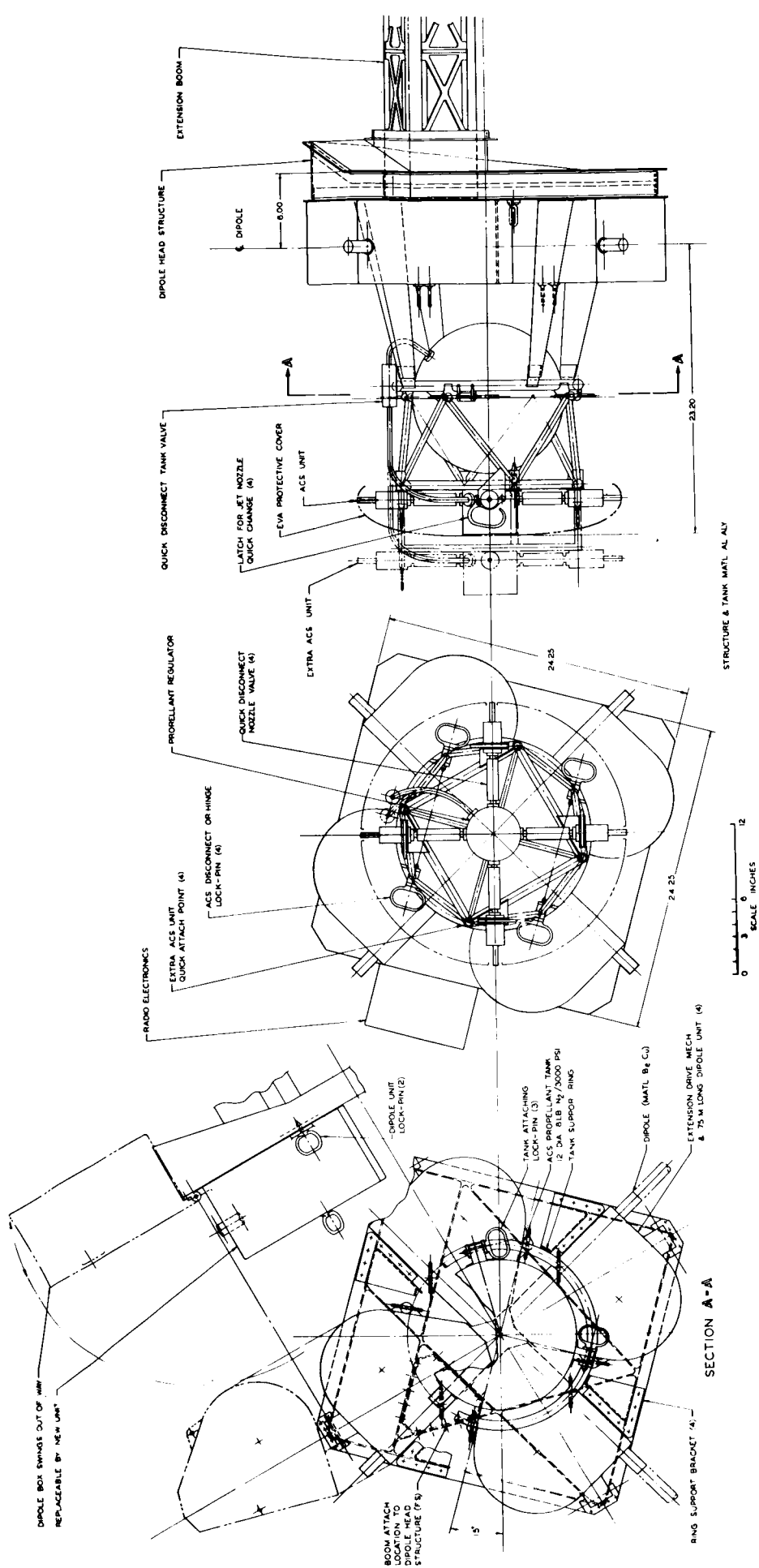
Figure 4-9. Dipole Head Assembly

The assembly is a replaceable module that is mounted with two pull pins and a spring loaded hinge. To replace it, the dipoles are retracted by remote command from the command module. The astronaut then pulls the two pins allowing the old assembly to rotate out of the way. He then receives the replacement assembly and an additional mounting pin from the SM by the clothesline or other means of transfer. He determines that the dipole is retracted to the same degree as the remaining three, then meshes the drive gear and presses in the pull pins and electrical quick disconnect.

The dipole itself, a wire mesh extendible tube, is stored as a strip on the spool in the dipole assembly and assumes its preformed shape of a 1.0-in.-dia. tube as it is extended. The mesh is 0.009-dia. wire at 12 wires to the inch. The longitudinal wires are Elgiloy to minimize the thermal coefficient of expansion and the circumferential wires are beryllium/copper to maximize the thermal conductivity. The entire mesh is silver plated and then painted black on the inside.

Antenna performance is dependent upon the precision of dipole length and position. The length is accurately determined and controlled by the linear counter. The position of the dipole is a function of dipole distortion caused by: 1) thermal gradients and the resultant unbalance of expansion and 2) cantelever bending due to dynamic forces. As indicated in Figure 4-10, the dynamic distortion is a direct function of the diameter and the thermal distortion is an inverse function.

The dynamic distortion curve in Figure 4-10 is derived from the curves shown in Section 4.4.3.2. The curves in Section 4.4.3.2 show the dipole tip deflection under the continued application of forces until the dipole assumes maximum theoretical deflection. From practical considerations, it is assumed that the maximum deflection to be expected from dynamic forces is approximately 1/4 the values of the theoretical maximum. The applied force is assumed to be 0.2 lb, obtained from the simultaneous firing of two jet nozzles. These estimated values are plotted as the appropriate dynamic deflection curve.

The maximum dipole tip deflection angle for acceptable rf performance is 5°. The analysis of two combinations of dipole materials; Invar-beryllium/copper and Elgiloy-beryllium/copper is shown in Section 4.5. Whereas both material combinations were found to be acceptable, the Elgiloy-beryllium/copper was used since it exhibited lower deflections and improved rf performance.

The thermal distortion curve in Figure 4-10 is derived from the curves shown in Figure 4-37. The two curves for the most detrimental condition of 10° solar angle assume that the tube seam is at 70° to the sun, the most detrimental position, or that there is no seam, the optimum condition. Again from practical considerations it is assumed that a seamed tube such as is used will rotate several times in 75m causing the seam to spiral along the dipole length. It is further assumed that this spiraling will result in a distortion that is mid-way between those of the no-seam and the 70°-seam condition. These estimated values are plotted as the appropriate thermal deflection curve.

NOTE: BASED ON 75 METER DIPOLE EXTENSION

BASED ON .009 ELGILOY LONGITUDINAL
MEMBERS AND BERYLLIUM/COPPER RADIAL
MEMBERS. 12 MEMBERS PER INCH.
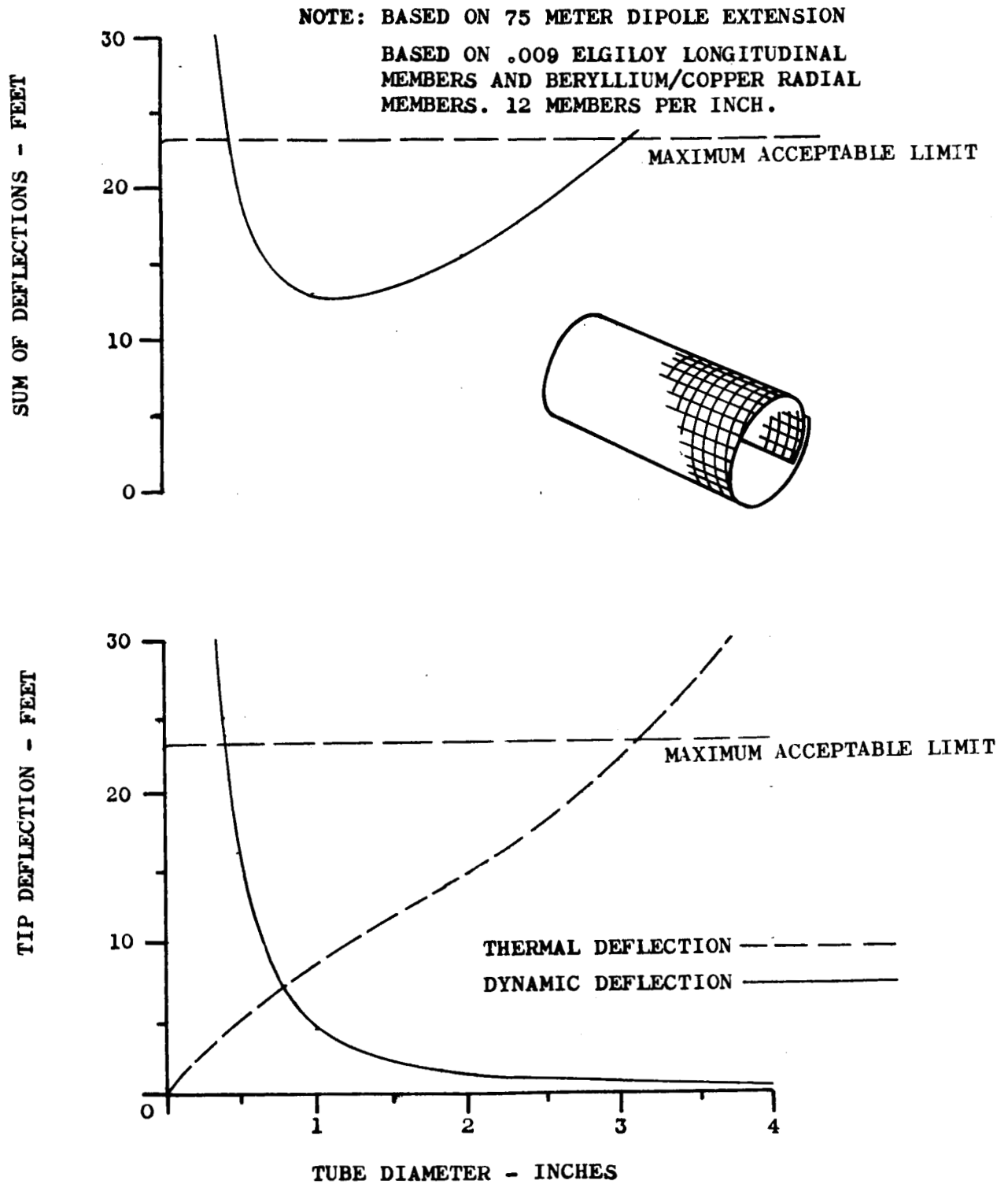
Figure 4-10. Dipole Deflection vs. Tube Diameter

4-26

The diameter for the mesh dipole is determined by the sum of the deflections. The combined tip deflection (maximum) at the chosen 1.00 in. diameter is approximately 13 in. in the 75m half length.

This deflection of the dipoles away from their nominal positions affects the performance of the array as an antenna. The deflection causes a shift in the effective position of the dipoles and also causes a change in the electrical impedance of the dipole. The measurement errors resulting from these deflections are complex and difficult to analyze quantitatively, but the magnitude of the deflections as shown by the curves is small. It is anticipated that a detailed analysis of the effects would prove the expected deflections to be tolerable for the precision required of the antenna system.

4.2.3 Stress Analysis

4.2.3.1 Ground Handling. All of the structural elements, including the dipole elements when stored on their drums are relatively rigid and no problems are anticipated during manufacture or ground handling of the retracted experiment.

4.2.3.2 Launch. Referring to Figure 4-5, the center-body structure consists of three mutually perpendicular octagonal frames joined to form a space truss. Bulkheads in the Y-Z planes (see Figure 4-11) provide shear rigidity in this plane and the retracted booms provide shear rigidity in the X-Z and X-Y planes. The entire structure is supported at points A, B, C, and D and weighs approximately 2500 lb. Maximum load factor = 4.9 limit (Ref. Volume II).

$$X_A = X_B = X_C = X_D = 0.25 \ (4.9) \ (1.25) \ (2500) = 3820 \text{ lb}$$

$$Z_A = Z_B = -Z_C = -Z_D = 7640 \left(\frac{36.2}{30}\right) = 9220 \text{ lb}$$
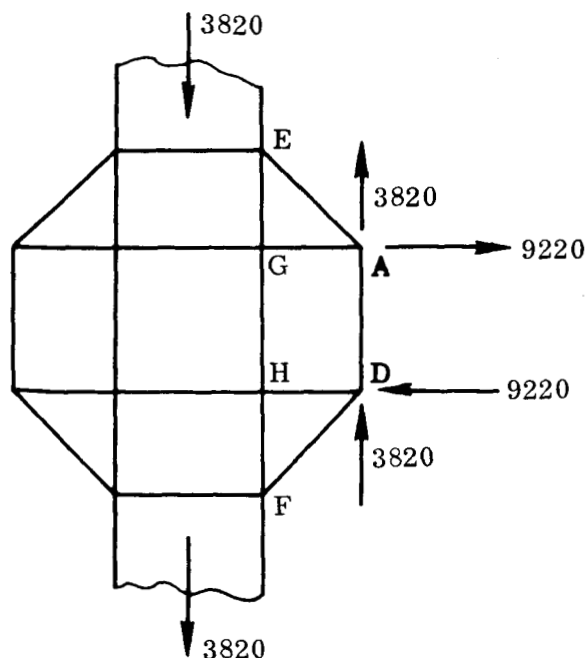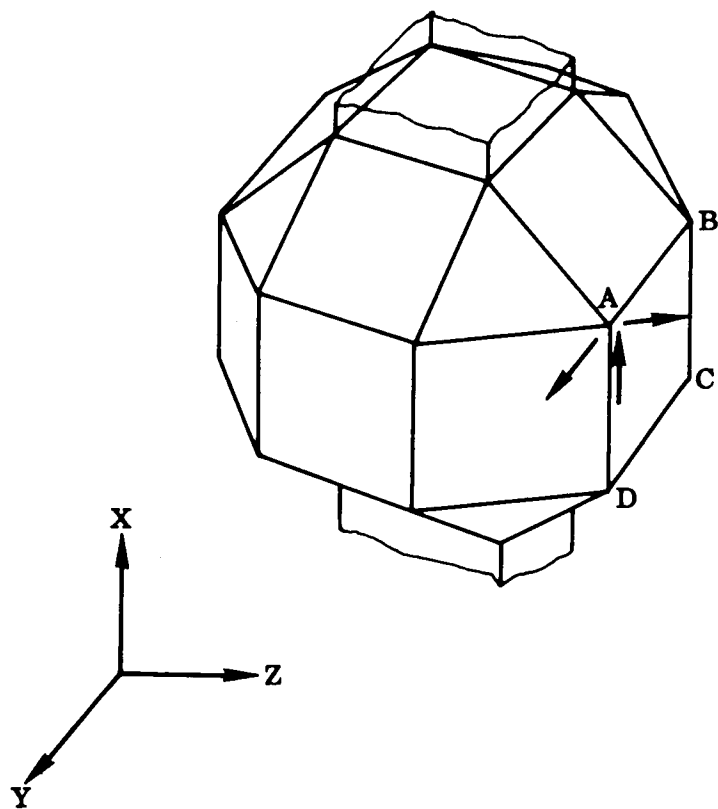
Figure 4-11. Planes and Axes of Center Body

$$P_{EA} = -P_{DF} = 1.414 \ (3820) = 5400 \text{ lb Compression}$$

$$P_{AG} = -P_{DH} = 9220 - 3820 = 5400 \text{ lb Tension}$$

$$A \approx 0.34 \text{ in.}^2$$

$$f_c = \frac{5400}{0.34} = 15,900 \text{ psi}$$
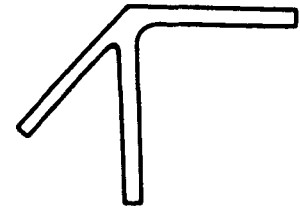
0.090 Extr.

Since column instability is precluded due to the support afforded by the solar cell panels, this stress is satisfactory.

Moment in boom $= 3820 \ (21.2) - 81,000 \text{ in.-lb}$

Shear in boom $= 3820 \text{ lb}$

Boom cap load $\dfrac{81,000 \ (0.707)}{36.4} = 1570 \text{ lb}$
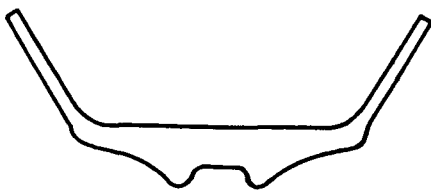
Cap area $\approx 0.3 \text{ in.}^2$
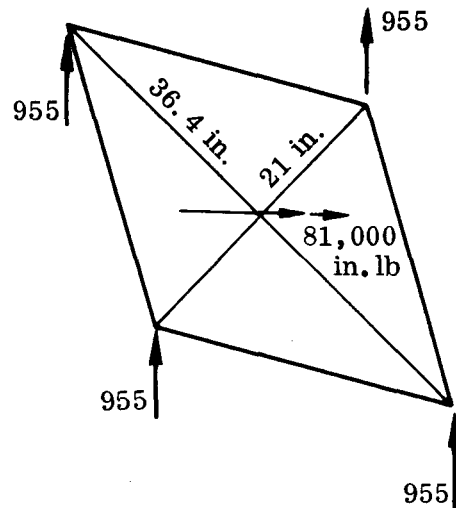
Conservatively assuming that 2 of the 8 nested caps carry the load:

$$f_c = \frac{1570}{0.6} = 2620 \text{ psi}$$

Shear in boom panel $= \dfrac{3820 \ (0.707)}{4 \ (0.866)} = 780 \text{ lb}$

This is the load in the upright members and is always compression if diagonal tension strap cross-bracing is used. The tension load in the diagonal is 1414 (780) = 1100 lb. Assuming 2 booms are effective:

$$f_t = \frac{1100}{2\,(0.5)\,(0.066)} = 17,200 \text{ psi (satisfactory)}$$

Assuming an upright area of 0.1 in.^{2}:

$$f_c = \frac{780}{2\,(0.1)} = 3900 \text{ psi}$$

While the bead-type uprights used for the outboard section are not adequate, it is obvious that a stable section can be substituted on the fixed sections in this region for a small weight penalty.

4.2.3.3 Operations. Critical loading on the boom occurs during a post-deployment maneuver. A docking load factor of 0.1 limit is assumed. The dipole head assembly was initially assumed to weigh 250 lb, and the boom weight varies linearly from 0.1 lb/in. at the top to 0.2 lb/in. at the center body.
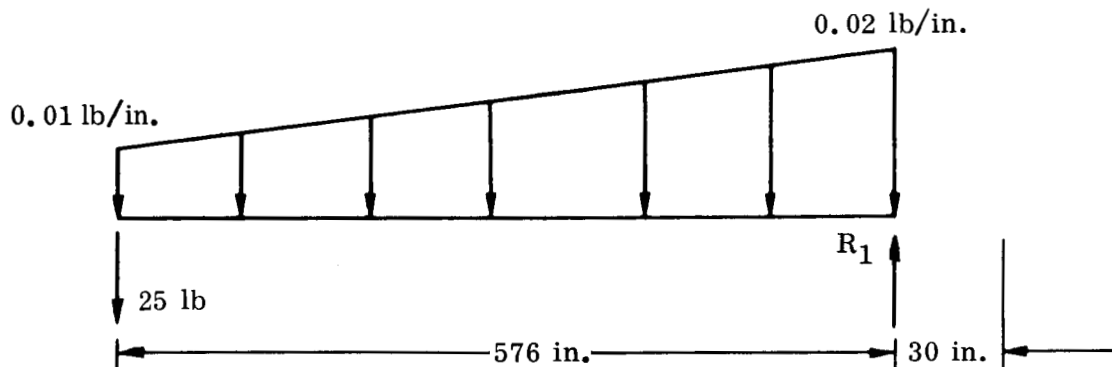
Max. moment $= 25\,(576) + 0.01\,(576)^2\,(0.5) + 0.01\,(576)^2\,(0.167)$

$= 16,600$ in.-lb limit

$$R_1 = \frac{16,600}{30} + 25 + 0.015\,(576)$$

$= 881$ lb limit

Cap load $= \dfrac{16,600}{16} = 1038$ lb
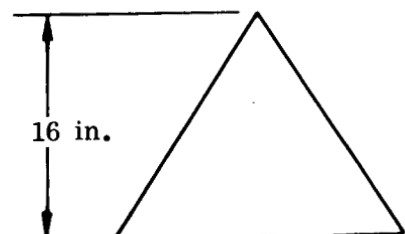
In addition to this axial load, some bending exists in the caps. Assuming a roller midway between uprights:

Roller load $= \dfrac{16,600}{37} + 25 + 0.015\,(576) = 495$ lb

$$M = \dfrac{495\,(12)}{4} = 1485 \text{ in.-lb}$$

A	y	Ay	Io
0.112	0.347	0.0388	0.0178
0.195	0	0	0.0004
0.307	0.347	0.0388	0.0182

$$\bar{y} = \dfrac{0.0388}{0.307} = 0.126$$

$$I = 0.0182 - 0.0049 = 0.0133$$

$$f_b = \dfrac{1485\,(0.568)}{0.0133} = 63,400 \text{ psi}$$

$$f_c = \dfrac{1038}{0.307} = 3400 \text{ psi}$$

$$f_{net} = 66,800 \text{ psi (limit)}$$
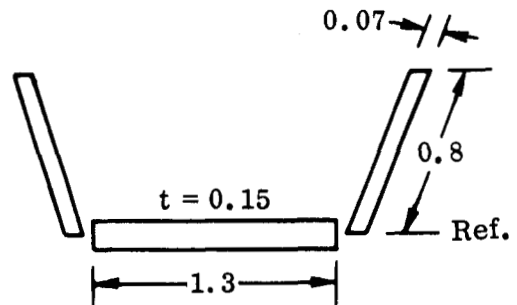
This stress is slightly excessive. However, it should be noted that the normal position of the rollers when the boom is completely extended is close to an upright and little bending exists. For the roller position analyzed, the boom must be partially retracted, and the boom moment is reduced. We conclude that the structure is feasible.

The load in the boom upright members is:

$$P = \dfrac{495}{2\,(0.866)} = 286 \text{ lb limit}$$

$$A \approx 0.1 \text{ in.}^2$$

$$f_c = \dfrac{286\,(1.25)}{0.1} = 3580 \text{ psi}$$

$$L/\rho \approx \dfrac{16}{0.1} = 160$$

$$F_c = \dfrac{\pi^2\,(E)}{(160)^2} = 3900 \text{ psi}$$

$$M.S. = \dfrac{3900}{3580} - 1 = \pm 0.09$$
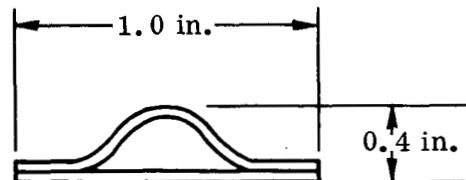
As an evaluation of the stiffness of the boom, the following calculation of tip deflection during a station-keeping maneuver is made. Two of the center-body jets are assumed to be operative, providing a net force of 0.2 lb. The total weight of the assembly is 4500 lb.

$$\text{Effective load factor} = \frac{0.2}{4500} = 0.0000445$$

$$\text{Tip force} = 0.000045\ (250) = 0.0111\ \text{lb}$$

$$\text{Distr. force} = 0.000045\ (0.1) = 4.5 \times 10^{-6}\ \text{lb/in. (tip)}$$

$$= 9.0 \times 10^{-6}\ \text{lb/in. (base)}$$

$$\frac{h_{TIP}}{h_{BASE}} = \frac{5.7}{18.5} = 0.308$$

$$I_{TIP} = 1/2\ (0.3)\ (5.7)^2 = 4.85$$

$$L = 591\ \text{in.}$$

Using the standard tapered beam deflection formula:

$$\delta = \frac{1}{EI_{TIP}}\ [0.05\ (0.0111)(591)^3 + 0.017\ (4.5 \times 10^{-6})(591)^4$$

$$+ 0.0042\ (4.5 \times 10^{-6}\ (591)^4]$$

$$= \frac{114,000 + 9000 + 2000}{4.85 \times 10^7} = 0.0026\ \text{in. tip deflection}$$

In view of this negligible deflection, the boom caps should be reviewed for possible weight reduction. It is to be noted, however, that the caps are primarily designed by local roller loads.

4.2.4 <u>Weights Summary</u>. The total weight of the crossed-H design is 4102 lb and it is almost equally divided between the upper and lower satellites. The lower satellite is slightly heavier than the upper due to additional radiometry correlation and telemetry equipment.

Care should be exercised in assigning electronic components to the satellites. Should one of these satellites become significantly heavier than the other, excessive amounts of propellant would be required to maintain attitude control.

A contingency and growth allowance of 10% has been added for the structural portions. To provide for redundancy, a 50% increase in the weight of the electronic components was also added to the contingency allowance (see Table 4-2).

The 4100-lb weight will easily meet the requirements of any of the Saturn V launch configurations being considered.

Table 4-2. Crossed-H Interferometer Weight Summary

	COMPONENT		WEIGHT (lb)	
	Complete Antenna			4101.8
	Upper Satellite		1684.4	
1-4	Boom & Mechanism	421.0		
5	Dipoles	122.8		
1-5	ACS (96 lb of propellant)	244.0		
5	Miscellaneous Antennas	9.0		
6	Solar Panels & Structure	234.5		
6	EVA Provisions	69.6		
7	Autopilot	57.5		
7	Tether Assembly	56.2		
7	Satellite Docking Structure	15.3		
8	CSM Docking Structure	127.5		
5	Electronics	202.0		
7	Battery & Charger	125.0		
	Lower Satellite		1751.4	
1-4	Boom & Mechanism	421.0		
5	Dipoles	122.8		
8	ACS (96 lb of propellant)	244.0		
5	Miscellaneous Antennas	9.0		
6	Solar Panels & Structure	234.5		
6	EVA Provisions	69.6		
7	Autopilot	57.5		
7	Tether Assembly	56.2		
7	Satellite Docking Structure	15.8		
8	CSM Docking Structure	127.5		
5	Electronics	269.0		
7	Battery & Charger	125.0		
	Contingency (10% Structure, 50% Electronics)		666.0	

4.3 <u>SUBSYSTEM DESIGN</u>. Subsystems design involves the following subsystems elements:

a. Radiometry receiving system.

b. Data transmission and telemetry.

c. Data processing.

d. Navigation and attitude control.

e. Power.

The functioning of each of these elements is closely related to that of each of the others. Major factors in the subsystems design are the assignment of functional characteristics for each element and the proper interfacing of all of the elements to obtain an optimum system performance. Some of the design characteristics and constraints on each of the subsystem elements are discussed in the following paragraphs.

4.3.1 <u>Radiometry Receiving System</u>. The radiometry receiving system is shown in simplified block diagram form in Figure 4-12. The principal elements of the system are the dipole antennas, the matching and phasing networks, the detectors and correlators and the timing and phasing system. Other elements are the radio-frequency amplifiers, filters, and the relay that provides for transmission of raw radiometry data from the upper satellite end to the lower satellite end of the interferometer.

The dipole antennas form the receptors for the radiometry system. On each satellite end of the interferometer are two sets of crossed dipoles comprising two end-fire arrays that receive orthogonally polarized radiant energy. The dipoles themselves are arranged so that in normal orientation of the satellite ends of the interferometer, they will project at $\pm 45^\circ$ angles with respect to the tether. This allows the ends of the interferometer to be tilted with respect to the tether without interference between the extended dipoles and the tether. Because of the 45° orientation of the dipoles, they will receive energy polarized in directions displaced 45° from the tether. To make processing of the received radiant energy data more convenient, the effective polarizations of the dipoles are rotated to be at 0° and 90° by means of the polarization networks described later on in this section.

The dipole end-fire arrays are designed to provide a cardioid-like radiation pattern, with enhanced reception in one hemisphere and greatly reduced reception in the opposite hemisphere, so as to remove ambiguities resulting from energy received from opposing directions. The radiation pattern of the arrays is given by:
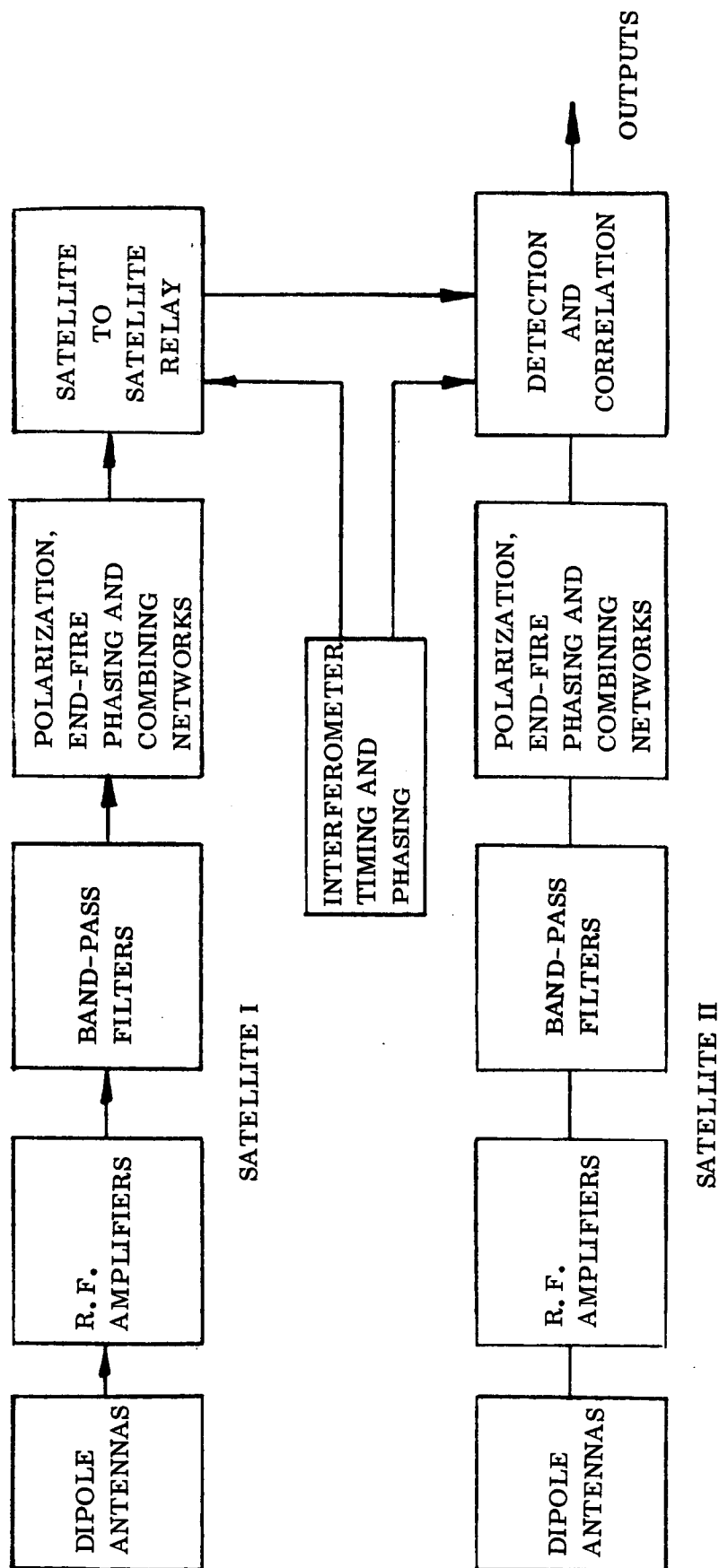
Figure 4-12. Radiometry Receiving System Simplified Block Diagram

$$E_\theta = \frac{\sin\left[\frac{\pi d}{\lambda}(\sin\theta\cos\phi + 1)\right]}{\sin\left(\frac{2\pi d}{\lambda}\right)} \frac{\left[\cos\left(\frac{\pi L}{\lambda}\cos\theta\right) - \cos\left(\frac{\pi L}{\lambda}\right)\right]}{\sin\theta\left[1 - \cos\left(\frac{\pi L}{\lambda}\right)\right]}$$

with $\frac{d}{\lambda} \le \frac{1}{4}$ and $\frac{L}{\lambda} \le \frac{5}{4}$

Orientation of the antennas in the coordinate system is illustrated in Figure 4-13.
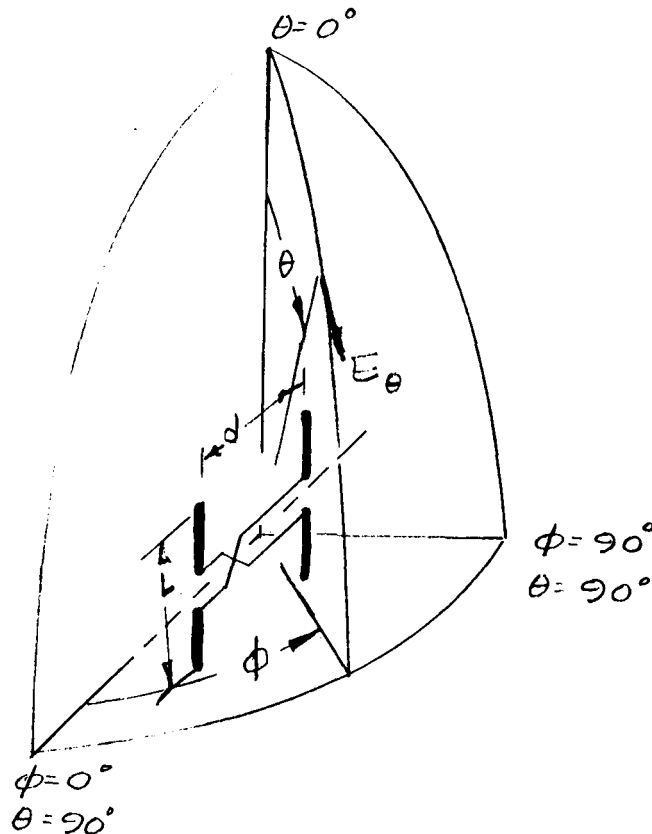
Figure 4-13. Dipole End-Fire Array

The lengths of the dipoles and the end-fire spacing between dipole sets are adjustable to permit operation over the complete 0.5 MHz to 10 MHz frequency range in three bands. These bands are tentatively set at 0.5 MHz to 2.5 MHz, 2.5 MHz to 5 MHz, and 5 MHz to 10 MHz. Dipole lengths and spacings for optimum end-fire radiation patterns are illustrated by Table 4-3.

Typical radiation patterns are shown in Figures 4-14 and 4-15. These patterns illustrate the range of variation in pattern shape over the entire 0.5 MHz to 10 MHz frequency band. In the 0.5 to 2.5 MHz range, the patterns vary as shown in Figure 4-15.

Table 4-3. Dipole Lengths and Spacings

FREQUENCY (MHz)	DIPOLE LENGTH L (meters)	DIPOLE SPACING d (meters)	$\dfrac{L}{\lambda}$	$\dfrac{d}{\lambda}$
0.5 ⎫ 2.5 ⎭	150	30	⎧ 1/4 ⎩ 5/4	1/20 1/4
2.5 ⎫ 5 ⎭	75	15	⎧ 5/8 ⎩ 5/4	1/8 1/4
5 ⎫ 10 ⎭	37.5	7.5	⎧ 5/8 ⎩ 5/4	1/8 1/4

Figure 4-14. End-Fire Array Beamwidth
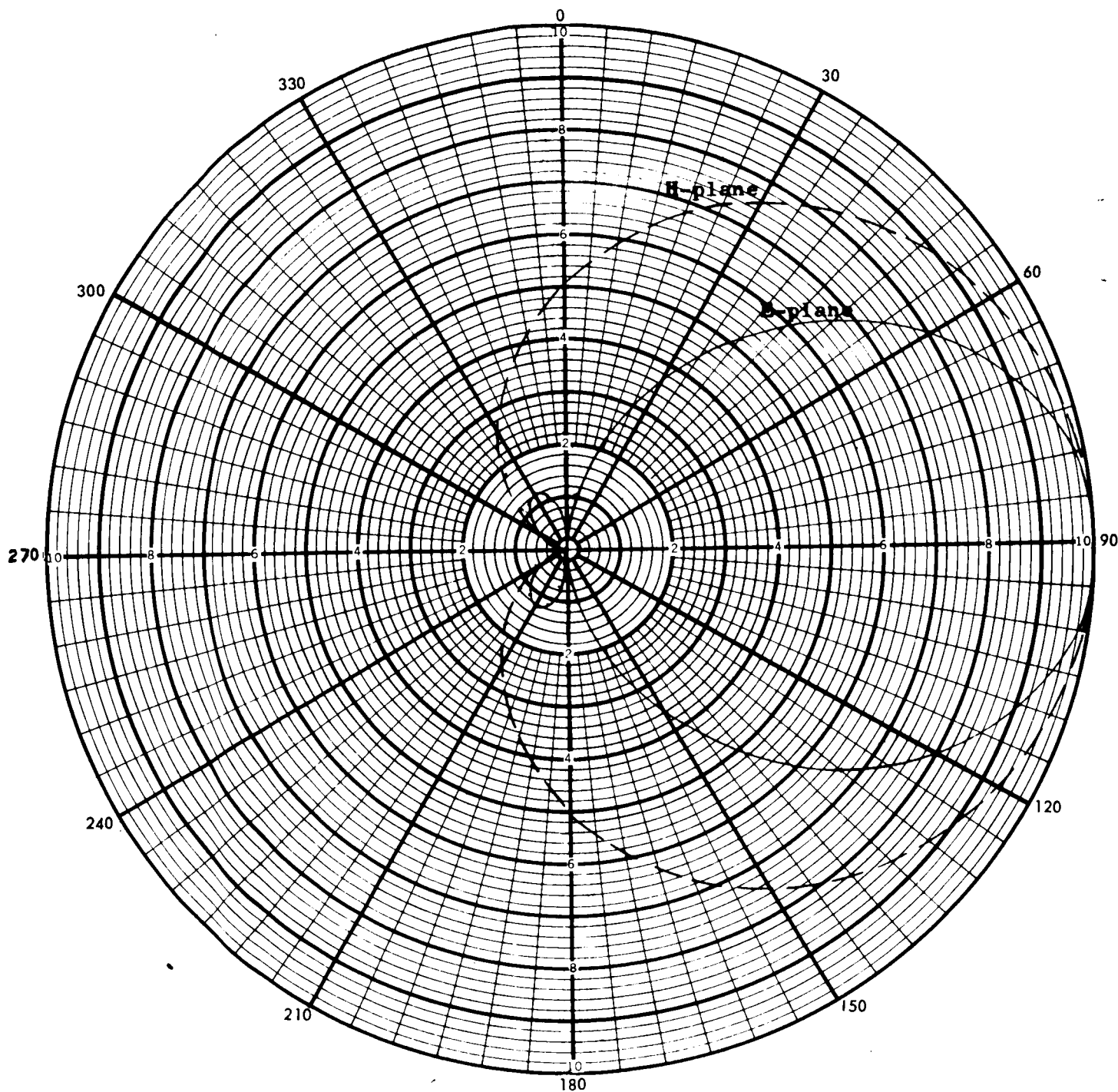
$\dfrac{d}{\lambda} = 0.05$, $\dfrac{L}{\lambda} = 0.25$, typical of 0.5 MHz

Figure 4-15. Radiation Patterns of End-Fire Array (Sheet 1 of 3)

4-38
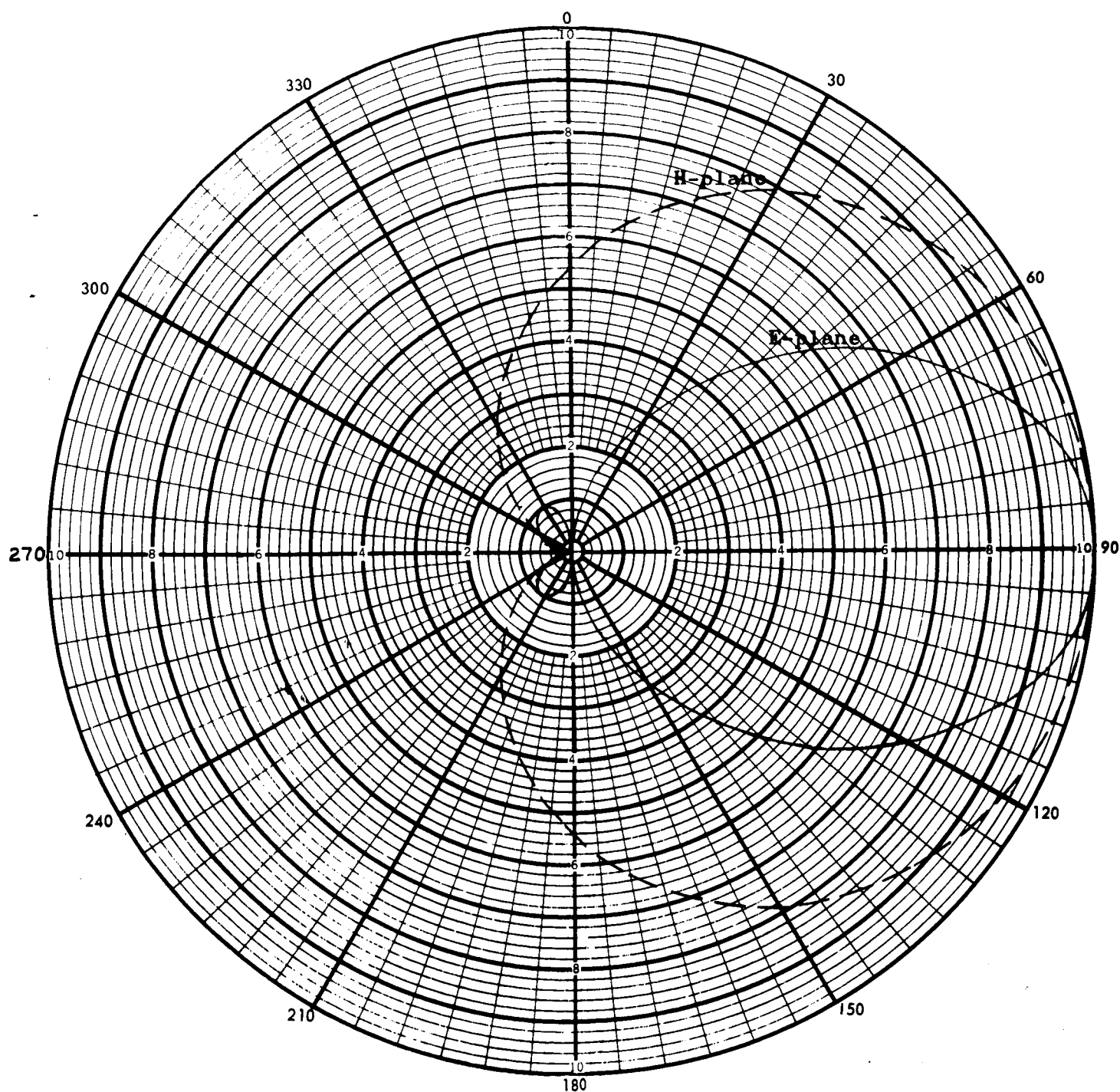
$$\frac{d}{\lambda} = 0.125, \quad \frac{L}{\lambda} = 0.625, \quad \text{typical of 1.25 MHz}$$

Figure 4-15. Radiation Patterns of End-Fire Array (Sheet 2 of 3)
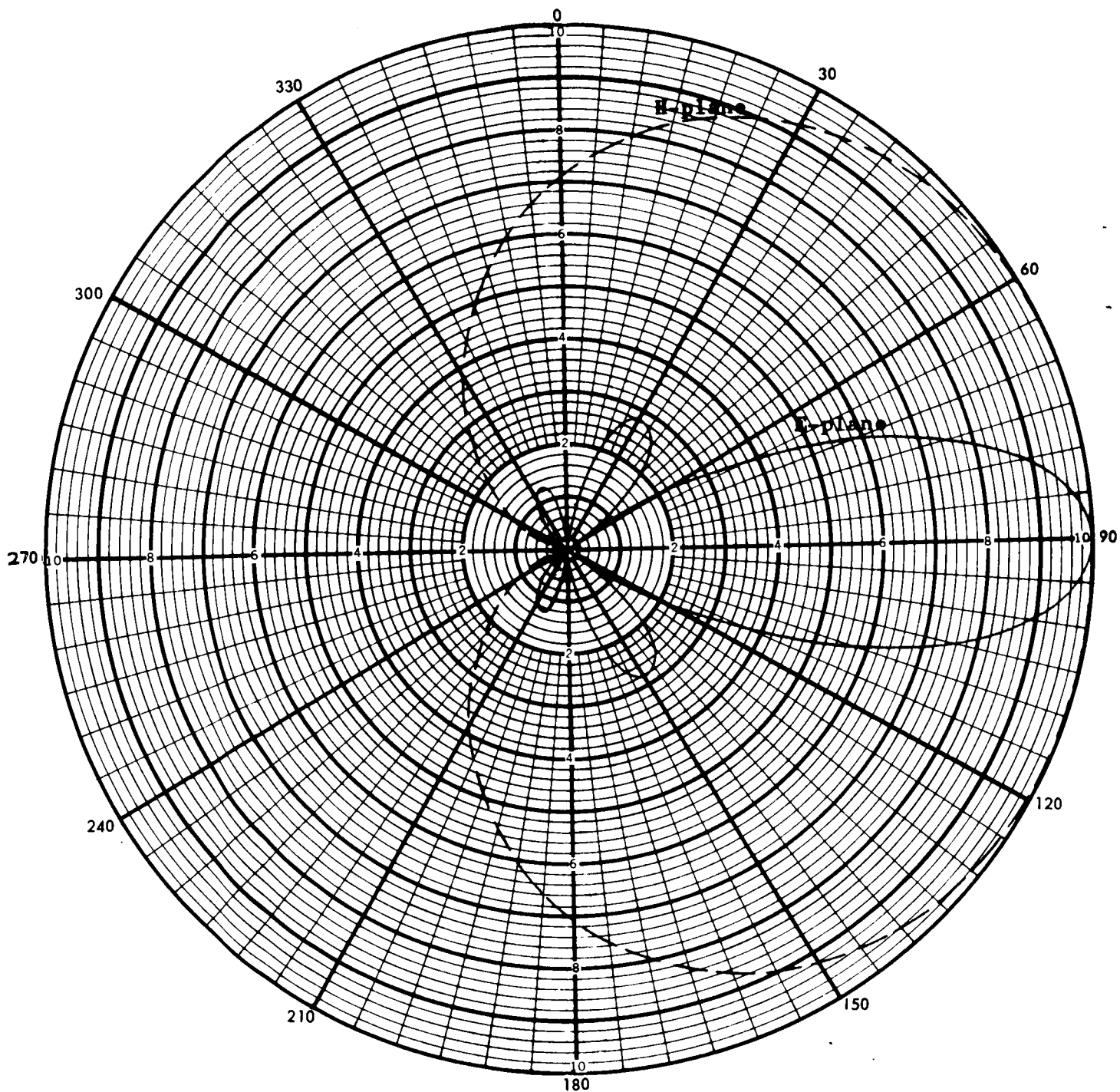
$\dfrac{d}{\lambda} = 0.25,\ \dfrac{L}{\lambda} = 1.25,$ typical of 2.5 MHz

Figure 4–15. Radiation Patterns of End–Fire Array (Sheet 3 of 3)

At 2.5 MHz and again at 5 MHz, the dipoles and their spacings are adjusted so that the pattern excursions are repeated.

The discrimination of the cardioid shaped patterns against reception from undesired directions to the rear of the array is evident from a review of the patterns. If we assign a -10 db reception level as a criterion, we find a conical region at least 106° wide wherein unwanted reception has been discriminated against. This is illustrated in another way by Figure 4-14, which shows the -3 db and -10 db beamwidths of the end-fire array as a function of frequency.

The discrimination effects just described enable the use of the interferometer to obtain valid measurement data without the ambiguity of simultaneous reception from opposite directions, in a conical region about the peak of the pattern that is at least 106° wide. Thus, with proper orientation of the end-fire arrays, we may map a complete sphere with good discrimination against ambiguous reception from opposing directions.

The method of providing the phasing necessary to obtain the desired end-fire radiation patterns is illustrated in Figure 4-16. Energy received at each dipole is amplified by a wide-band amplifier, then sent through band-pass filters that separate out different portions of the spectrum for transmission through different circuits. This separation into narrow frequency bands is necessary if good front-to-back ratios of reception are to be obtained throughout the frequency band with the end-fire arrays, because of the mutual impedance properties of the dipoles in the arrays. At each narrow frequency band the end-fire phasing components shown in the circuit of Figure 4-16 insert the phase shift and magnitude transformations required for proper cardioid pattern shape.

It will be noted that the circuit is designed to provide a dual set of cardioid patterns simultaneously; that is, through the use of a dual hybrid and phasing network arrangement, if reception (and null) of each is in a direction opposite to that of the other. Thus data from opposite hemispheres can be simultaneously obtained.

The band-pass filters also serve another function, that of limiting the spectral spread of any one set of observed data, so that phase shift techniques rather than more difficult transmission delay techniques can be used in the synthetic aperture correlation process. Filter pass-band widths on the order of 2 or 3 kHz appear to provide appropriate system performance. About 15 pass bands, i.e., five in each of the three principal divisions of the 0.5 MHz to 10 MHz frequency range, would give a reasonable spectral sampling for the interferometer mapping mission.

Not specifically shown in Figure 4-16, but a circuit feature that could be provided, would be servo-controlled filter and end-fire phasing elements that would permit swept-frequency measurements to be made. This capability is most desirable for the observation of strong time-varying sources.
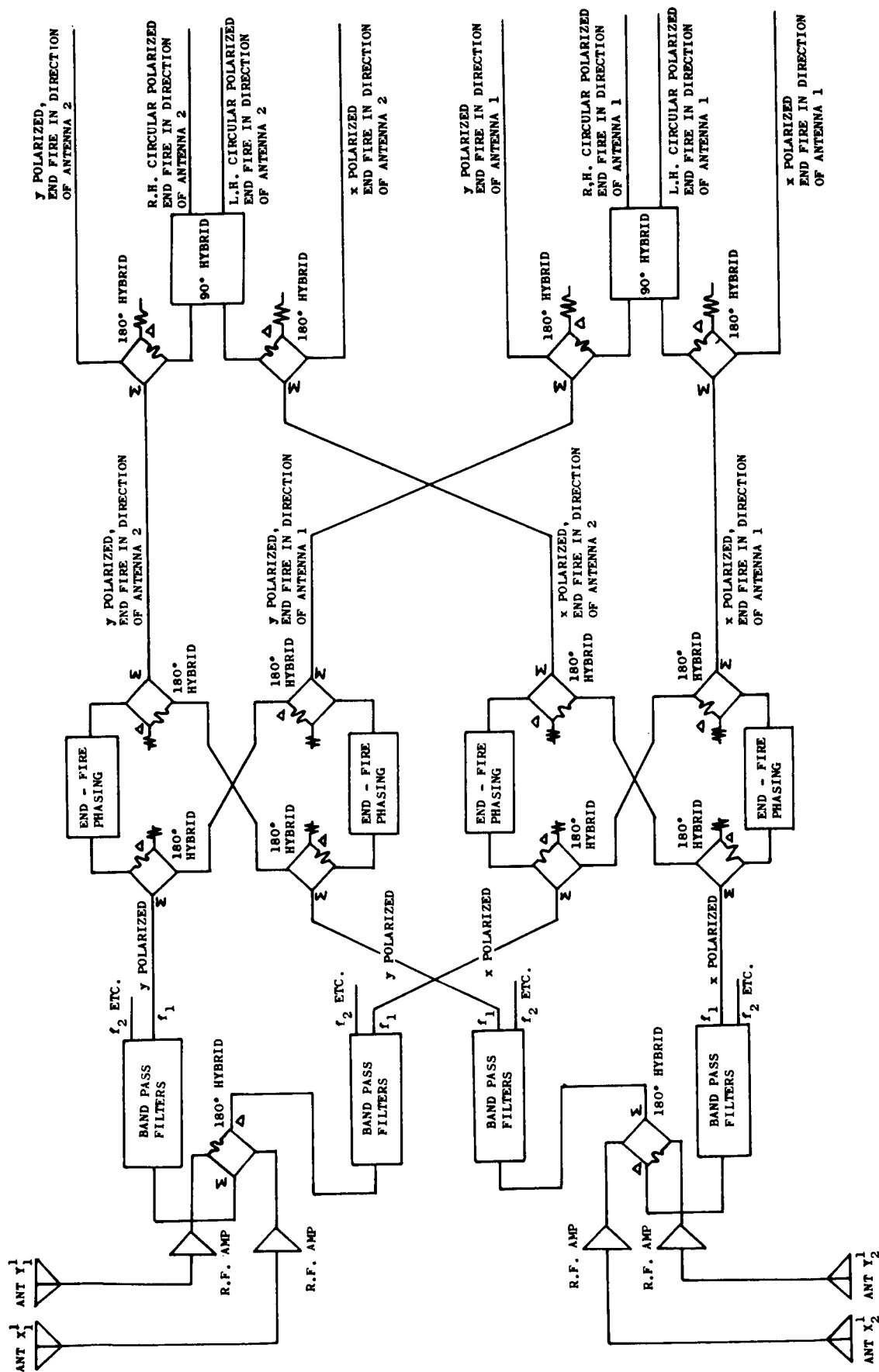
Figure 4-16. Typical Radiometer RF Phasing and Combining Circuit

For operation of the interferometer in the mapping mode, the configuration of the dipoles and circuits in each satellite are kept identical. For the observation of time-varying sources during which the interferometer function is not used, the configuration of the two satellites could be made different so that measurements in different frequency bands could be made simultaneously.

The phasing and combining circuit of Figure 4-16 also provides for the simultaneous measurement of various polarization components of the received radiant energy. Typically, two orthogonal linear polarization components and right-hand and left-hand circular polarization components of the received energy would be measured. From these measurements the complete polarization characteristics of the energy could be determined.

The outputs of the phasing and combining circuit, shown at the right of Figure 4-16 lead to the detection and correlation portions of the radiometer equipment. Envelope or power detectors would give a measure of the energy incident on each of the channels of the circuit of Figure 4-16. Correlation or product detectors, measuring the correlation between inputs from the two ends of the interferometer, would yield values of the Fourier components of the sky spatial radiant distribution, which later could be processed through ground-based computers to obtain maps of the sky brightness distribution. A representative correlation detector, of a type developed by Hubbard and Erickson (Reference 5), is illustrated in Figure 4-17.

4.3.2 <u>Data Transmission and Telemetry</u>. The functioning of the crossed-H interferometer requires a considerable transmission of data and commands between the ground and the interferometer and between satellites of the interferometer. The simplified information flow chart of Figure 4-18 illustrates the general nature of the required information flow.

The Satellite II end of the interferometer is normally the one nearest the earth when deployed in orbit. It functions as the headquarters for the interferometer, with the Satellite I end of the interferometer acting as a semi-independent unit, although for a final design a redundancy of equipment likely would be provided so that either satellite could operate as headquarters on command, for greater reliability.

Information flowing from the ground includes commands for orienting the interferometer in its orbit and for adjusting the dipoles and boom. It also includes commands for tuning and operating the radiometry instrumentation equipment. It is likely that a time reference signal would also be transmitted to the interferometer to up-date time signals generated at the interferometer.

Information flowing between the satellite ends of the interferometer includes command signals transmitted from the headquarters Satellite II to Satellite I and information flowing the opposite way on the status of instrumentation, of satellite configuration and orientation, and of other equipment aboard Satellite I.

Figure 4-17 Stabilized Cross-Correlation Detector
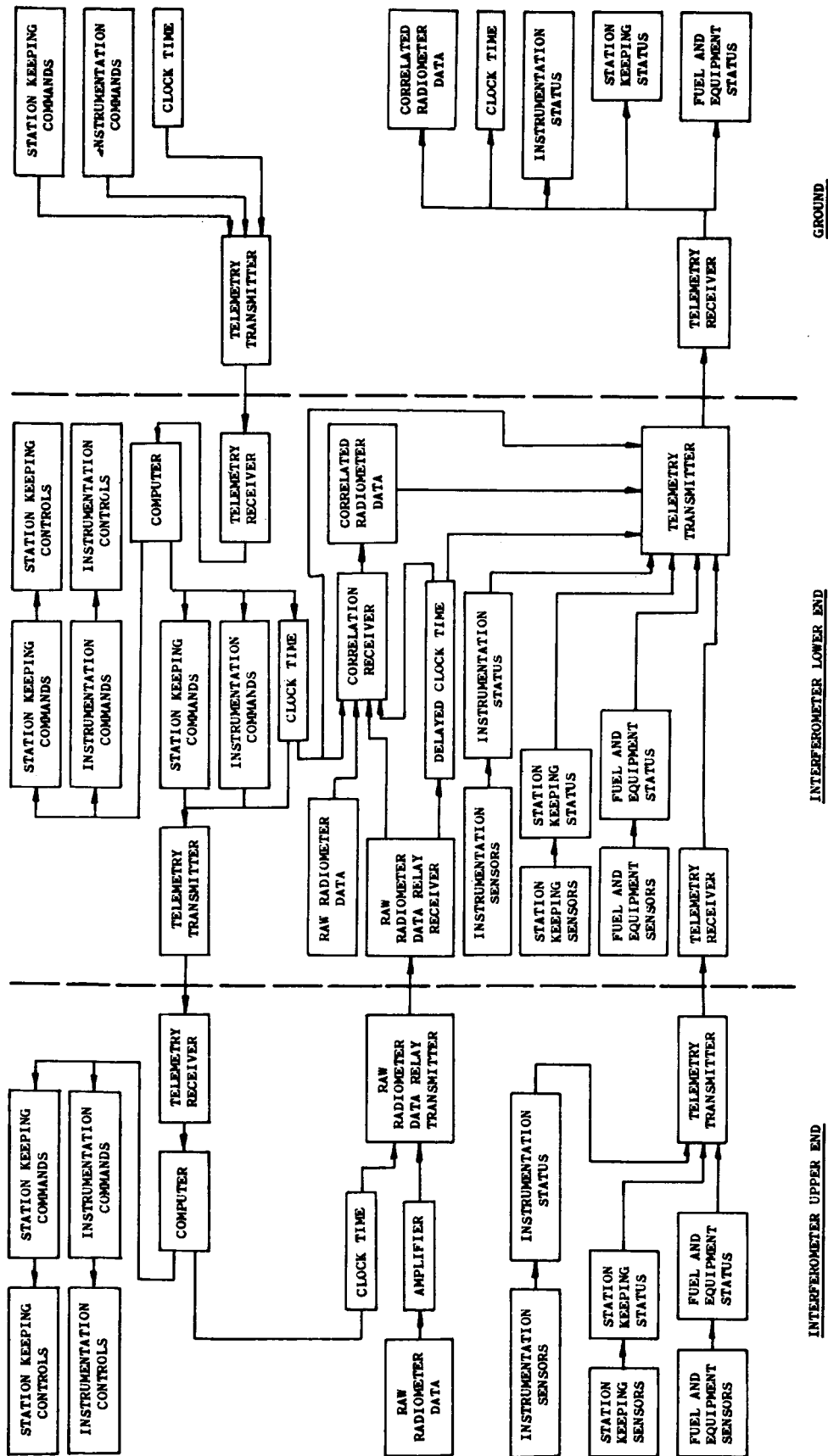
Figure 4-18. Simplified Information Flow Chart

4-45

Raw radiometry data are also relayed from Satellite I to Satellite II, where it is partly processed and correlated with other raw radiometry data received directly at Satellite II, before being transmitted, at reduced data rates, to earth for further processing.

A consideration in the operation and use of the interferometer is an accurate knowledge and control of the phasing and timing of radiometry data received at the two ends of the interferometer, in order that the correlation and processing of the data will yield useful results. A high data rate transfer of timing signals, together with highly accurate data on the relative positioning of the two ends of the interferometer, is required between the satellites.

Data transferred from the interferometer to ground include partly processed radiometry data, time reference signals, and data on the status of interferometer instrumentation, orientation, and equipment.

Data rates between satellites must be high to satisfy the demands for timing information and for unprocessed experimental data transfer. In these links the data rate requirements are about 5×10^6 bits per second. For the earth-interferometer links the rates are much lower, being on the order of 10 kilobits per second. This lower data rate is possible because of the partial processing of radiometry data at the interferometer, which also reduces the power requirements for data transmission to reasonable values.

A representative equipment system to perform the information flow add data processing functions of the interferometer is shown in the simplified equipment block diagram, Figure 4-19. While the final definition of experiment equipment will be made by the various principal investigators, it is expected to be quite similar to that illustrated. Physical characteristics of the radiometry equipment are summarized in Table 4-4. Relative location of components is shown in Figure 4-20.

4.3.3 Navigation and Attitude Control System. This subsystem performs the instrumentation and computing tasks for both the navigation and the attitude control functions. The integration is discussed in Section 4.3.3.1 and the subsystem is selected in 4.3.3.2. Operation is examined in 4.3.3.3 and data processing considerations are in 4.3.3.4. Finally, specifications are presented.

4.3.3.1 Integrated Navigation and Attitude Control Functions. The instrumentation and computing tasks of the navigation function and the attitude control function is integrated to avoid equipment duplicity. Navigation is concerned with determination of the satellites' position and velocity, expressed in the desired coordinate system. In general, the navigation function provides:

a. Orbit determination — estimation of orbit by means of navigational fixes with/without ground tracking.

b. Orbit integration — continuous estimate of satellites' orbital position and velocity.
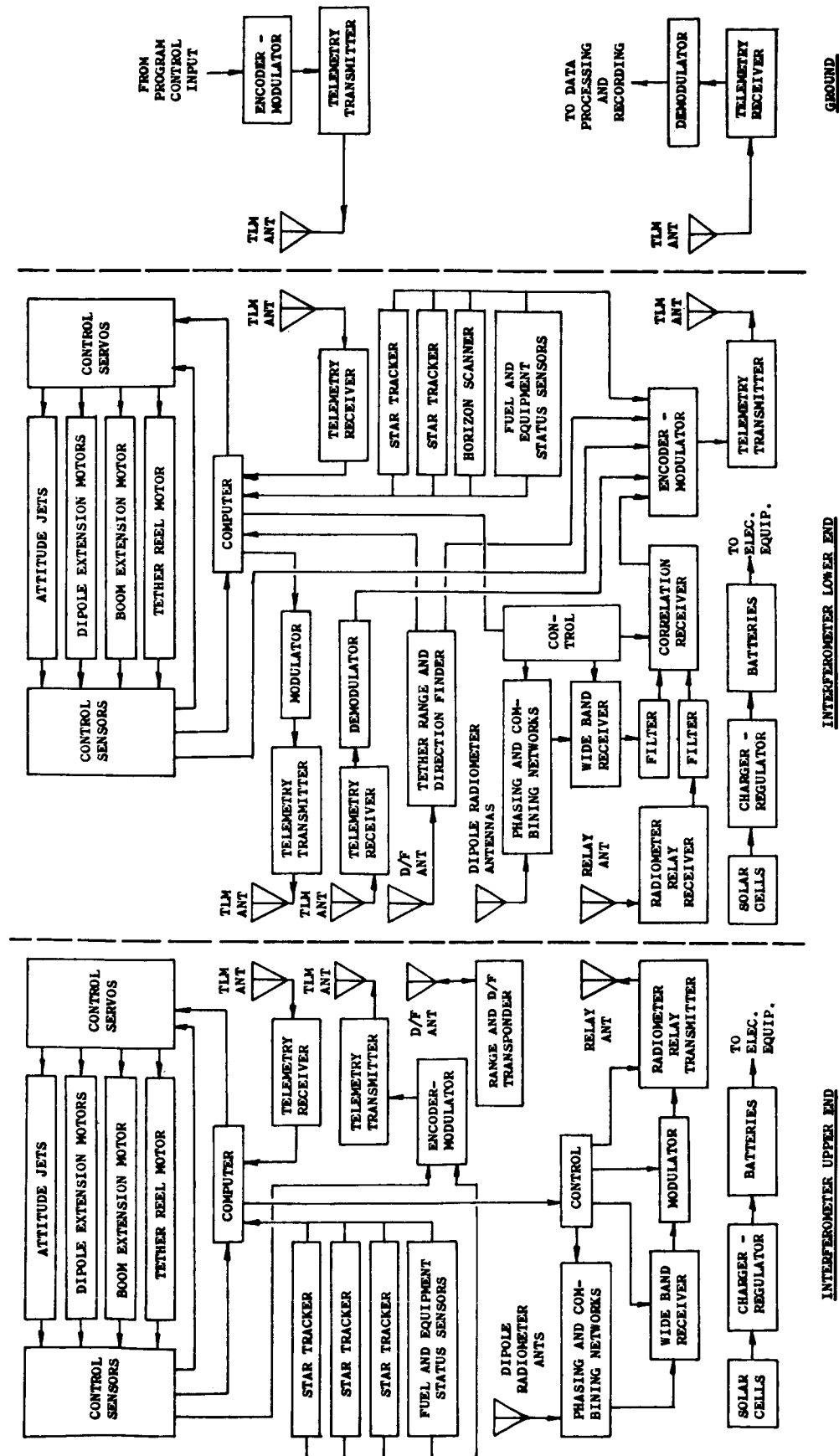
Figure 4-19. Simplified Equipment Block Diagram

4-47

Table 4-4. Assumed Typical Experiment Equipment List

EQUIPMENT	SIZE (ft³)	WEIGHT (lb)	POWER (watts)
SATELLITE I			
Range & D/F Equipment	0.17	10	2
Computer (For instrumentation only)	0.5	50	50
Relay Receiver	0.06	4	2
Phasing & Combining Networks	0.25	20	1
Equipment Sensors	1	10	5
Wide-Band Radiometry Receivers (8 each)	0.5	30	20
Swept Frequency Radiometry Receivers (3 each)	0.5	25	20
Radiometry Filters (120 each)	1	40	1
Radiometry Power Detectors (120 each)	0.3	20	15
Radiometry Correlation Detectors (40 each)	1	60	50
TOTALS:	5.28	269	186
SATELLITE II			
Range & D/F Transponder	0.04	3	2
Computer (For instrumentation only)	0.5	50	50
Relay Transmitter	0.06	4	15
Phasing & Combining Networks	0.25	20	1
Equipment Sensors	1	10	5
Wide-Band Radiometry Receivers (8 each)	0.5	30	20
Swept Frequency Radiometry Receivers (3 each)	0.5	25	20
Radiometer Filters (120 each)	1	40	1
Radiometer Power Detectors (120 each)	0.3	20	15
TOTALS:	4.15	202	129

ANTENNAS

1, 2 TLM SC/SC Rec'v.
2" Whipes Opp. Sides

3 TLM SC/SC X'mit.
4" Dia. Spiral

4 Range & D/F & Relay
Rec'v. & X'mit. Plane
Array 12"x'2"x1"

5 Docking Beacon,
Whip on Docking Ring
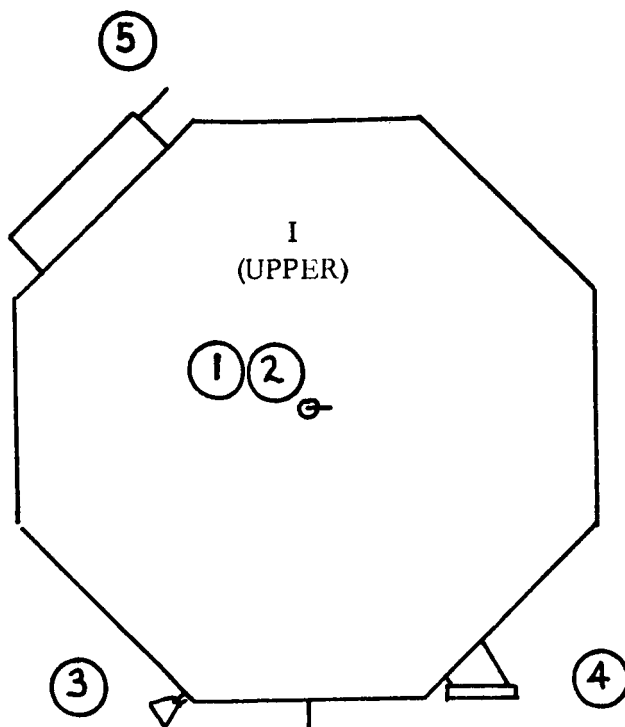
6, 7 TLM SC/SC Rec'v. &
TLM GND/SC Rec'v.
2" Whips Opp. Sides

8 Range & D/F & Relay
Rec'v. & X mit. Plans
Array 12"x12"x1"

9 TLM SC/SC X'mit.
4" Dia. Spiral

10 Docking Beacon,
Whip on Docking Ring

11 TLM SC/GND. X'mit.
4" Dia. Spiral
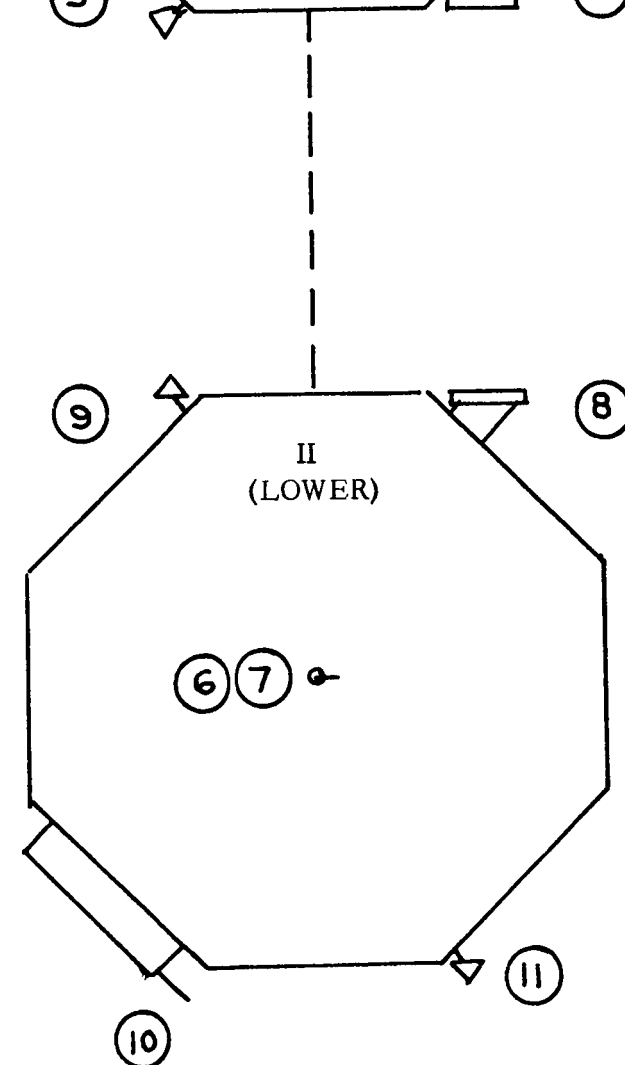
Figure 4-20. Data Transmission and Telemetering Antennas

Attitude control requires:

a.　Attitude determination — orientation of satellites.

b.　Rate and acceleration of attitude.

Additional functions include:

a.　Failure detection logic.

b.　Control of the operational phases.

c.　Obtaining minimum fuel and power expenditures under all operational phases.

d.　Self-program and interface with TLM (ground-track) program.

e.　Backup capability.

4.3.3.2 Selection of Navigation-Attitude Control Subsystem.   Possible instrument combinations and computational options are shown in Table 4-5. A semi-autonomous system is preferred with a ground tracking system as backup.

The subsystem, showing the major components and sensors for a semi-autonomous approach is shown in Figure 4-21 for Satellite II, which is the lower satellite. The two satellites have identical equipment, except that the lower satellite has a horizon scanner and a star tracker while the upper satellite has two star trackers. A GIMU (strap-down gyros or gimballess inertial measurement unit) was placed on both satellites following considerations of weight, power, volume, reliability, and maintainability. The additional computation required to mechanize the GIMU, while by no means trivial, is not a great price to pay for removing the gimbals and platform. Local-vertical information can be updated by analytical techniques, since it can be analytically derived.

The digital computer in the figure handles computational tasks for both navigation and attitude control functions. For example, one task is the navigation function of providing the appropriate coordinate systems and the necessary transformations. An attitude control function is due to the selected instrumentation including neither rate nor acceleration sensors. Therefore, rate, acceleration, and possibly rate of change of acceleration, may be derived or predicted.

The preferred system is a semi-autonomous system; the backup system is a ground tracking system as indicated in Table 4-5. On a computation and reliability point of view, the ground-track system would be selected. Ground-based tracking networks and computers provide unlimited computational capability. On that premise, we could treat the navigation problem as a data filtering problem or a discrete optimal filtering problem. On a completely autonomous system, the requirements would impose tremendous computer loading, an autonomous system implies complete automation and more complex on-board electronics. However, navigational autonomy for simultaneously

Table 4-5. Navigational–Attitude Control Instrumentation and Computational Options

OPTIONS	LOCATION		COMPUTATIONAL OPTIONS		
	SATELLITE I (UPPER)	SATELLITE II (LOWER)	AUTONOMOUS (ON-BOARD NAVIGATION COMPUTATION)	SEMI-AUTONOMOUS (ON-BOARD/GROUND TRACKING NAVIGA-TIONAL COMPUTATION)	GROUND TRACKING NAVIGATION COMPUTATION
1. Two Star Trackers *Inertial Platform (3 gyros)	X				
2. One Star Tracker, Inertial Platform (3 gyros)	X				Backup
3. Two Star Trackers, GIMU (Strapdown 3 gyros)	X			Preferred	
4. One Star Tracker, GIMU (Strapdown 3 gyros)	X				
5. One Star Tracker, Horizon Seeker, Inertial Platform		X			
6. One Star Tracker, Horizon Seeker, GIMU (Strapdown 3 gyros)		X		Preferred	Backup

Notes: 1. With redundancy in some areas, components will differ from above listing.
2. Purpose of inertial platform is to provide a "memory" for local vertical for Satellite I; alternate is a continuous analytical determination.

*Unless relationship between booms is known.
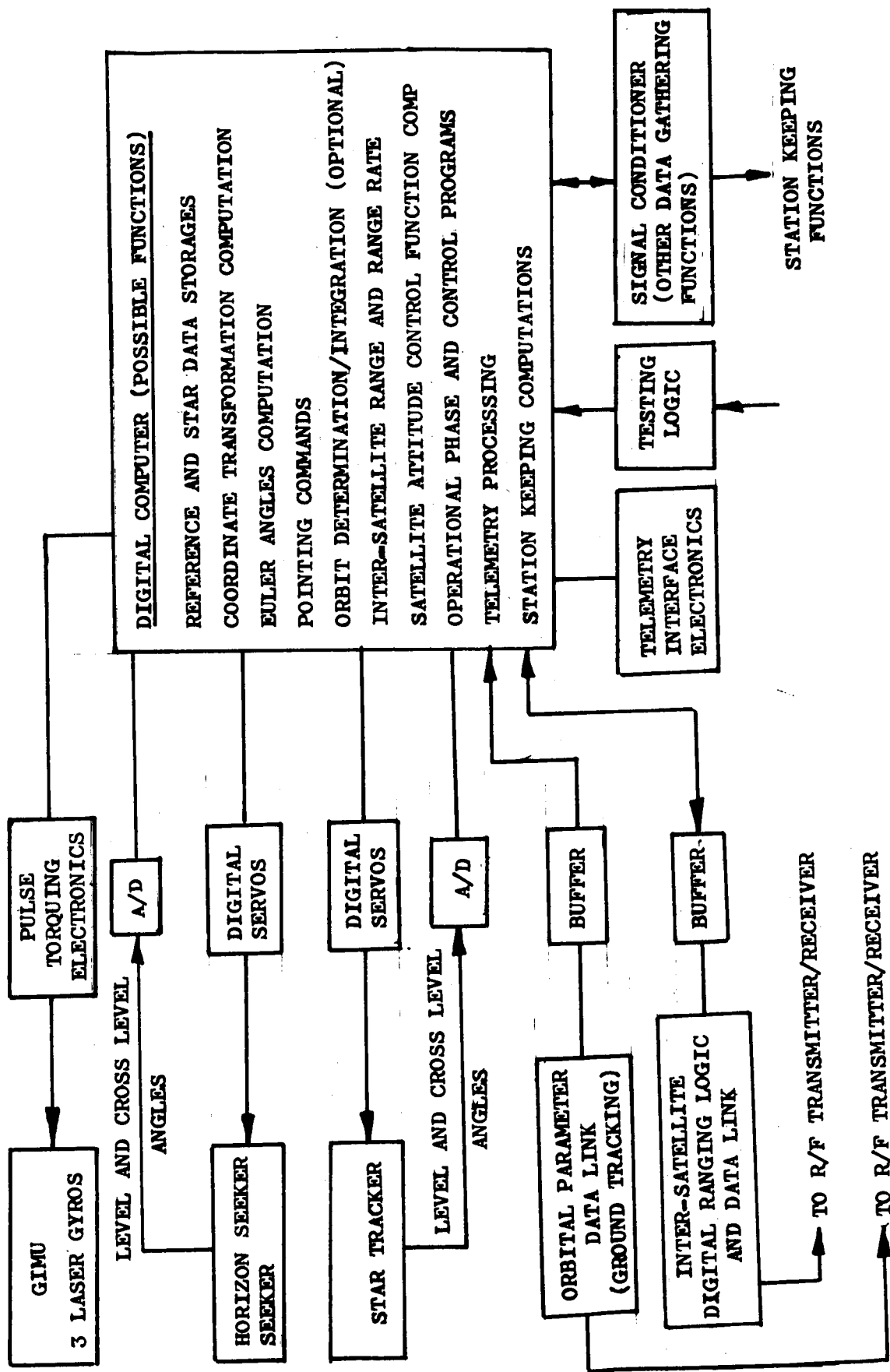
Figure 4-21. On-Board Semi-autonomous Navigation Attitude Control
Subsystem Equipment Diagram for Lower Satellite

orbiting satellites is desired since the ground tracking facilities may be assigned to other missions.

The desired combination is a semi-autonomous system. In this system, the primary tasks of the ground tracking network are:

a. Orbit determination.

b. Orbit integration.

c. Program definition.

The on-board instrumentation then will provide functions such as pointing command computations or automatic attitude determination as discussed later.

4.3.3.3 Operation

Attitude Determination.   Since the fixed stars provide true inertial reference, it is possible to deploy gimbaled vehicle-mounted star trackers for attitude determination by establishing LOS (line of sight) directions to various stars in vehicle coordinates. Given the orbital parameters and time, in general we may obtain satellite attitude by using a single star tracker and a horizon seeker and gyros that are either platform-mounted or strapped-down (GIMU) or by using two star trackers.

Initial Data and Orbital Parameters.   Given the coordinate system, usually an ecliptic or equatorial coordinate system, reference star data may be stored in terms of the components of a pointing vector. (Data are stored in the computer.) Transformation of this pointing vector into a coordinate frame defined by the orbital plane and position can be accomplished by a transformation matrix whose elements are direction cosines, which are transmitted from ground tracking stations.

Alternatively, we may transform stored data in body coordinates by a rotation vector. The rotation vector is used to orient the vehicle axes to some desired orientation.

Operational Phases.   The operational phases are indicated in Figure 4-22. Note that the upper satellite has no horizon seeker, which means local-vertical must be computed. However, initial local-vertical data (prior to satellites separation) will be transferred to Satellite I from II, and stored. Updating can be accomplished thereafter by star reference or by radar reference to the lower satellite.

Initial Coarse Alignment.   The initial coarse alignment is achieved by:

a. Alignment of satellites with the orbital plan and uncaging to this attitude.

b. Slaving the satellites to the gimbaled horizon seeker after azimuth alignment.

Figure 4-22. Operational Phase Modes
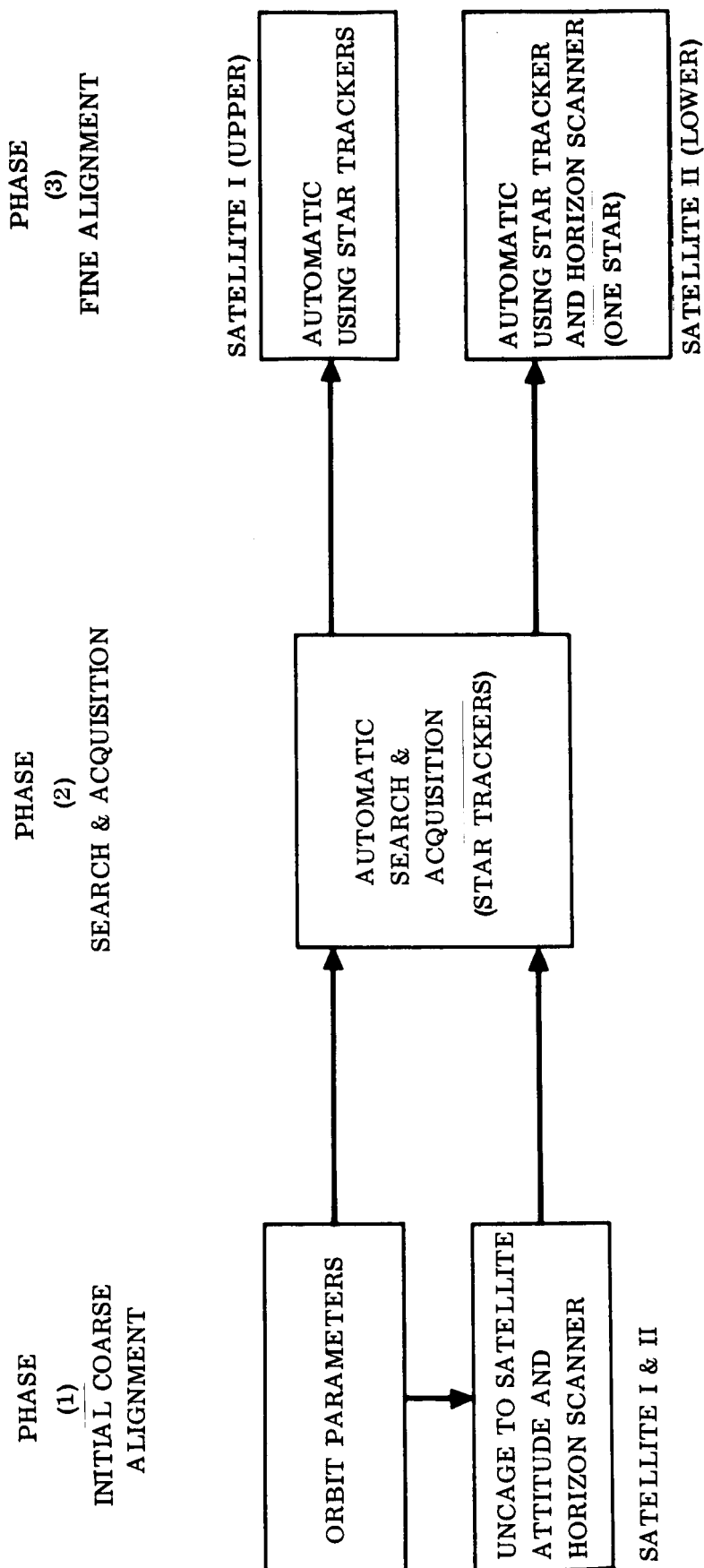
NOTES: UNCAGE TO SATELLITE ATTITUDE ≈ BOOSTER ATTITUDE PRIOR TO EJECTION.

PHASE (1) INITIAL COARSE ALIGNMENT

PHASE (2) SEARCH & ACQUISITION

PHASE (3) FINE ALIGNMENT

ORBIT PARAMETERS

UNCAGE TO SATELLITE ATTITUDE AND HORIZON SCANNER

SATELLITE I & II

AUTOMATIC SEARCH & ACQUISITION (STAR TRACKERS)

AUTOMATIC USING STAR TRACKERS

SATELLITE I (UPPER)

AUTOMATIC USING STAR TRACKER AND HORIZON SCANNER (ONE STAR)

SATELLITE II (LOWER)

In either case, we assume the CSM is properly aligned to the orbital plane, in which case the gyros are uncaged prior to satellites final ejection. Therefore, the coarse alignment problem is simply the determination of the error signals that will orient the CSM in relation to the orbital velocity vector, while the horizon seeker provides local vertical orientation for both satellites, which are connected.

Search and Acquisition. Star acquisition is accomplished by pointing commands, computed or provided by ground link, which initiates a search mode followed by lock-on. The pointing commands are resolved in body coordinates and then used to provide servo control signals for gimbal drives to the star tracker. The problem is how to compute the designed pointing commands. Since one satellite (lower) uses a horizon seeker and a single star tracker, and the upper satellite uses two stars or two trackers, analytical determination of local vertical for the upper satellite must be done. Therefore, the solution of the level and cross local gimbal angles of the star trackers for both satellites must be obtained. The procedure is to first transform star reference data from inertial to orbital coordinates. Next, this vector is resolved in body coordinates and we solve for the star tracker gimbal angles. The solution of these angles is the pointing commands. Once a star reference has been acquired, the tracker error signals are employed for tracker gimbal control in place of the initial pointing commands.

An alternative to above procedure is to compute required pointing commands in a ground station.

Fine Alignment. Fine alignment is the terminology used for orienting the satellite in a desired position with respect to inertial space.

For the upper satellite, we could have a star tracker and observe two stars sequentially or, as stated earlier, we could use two star trackers.

The observed (2 stars) LOS's expressed into orbital coordinates, is compared to the stored computed star LOS in orbital coordinates. The satellite is rotated until the two vectors are coincident. Alternatively, we may solve for the rotation vector.

4.3.3.4 Data Processing Considerations. The digital data link provides a digital communication link between satellites and between the ground tracking network and satellites. Data are processed in the digital computer when required. Included in the data link electronics are error detection logic and appropriate buffer for synchronization and/or conditioning.

Typical data transmitted would be orbit parameters, satellite attitude, range and range rate between satellites, commands for attitude control, and pointing commands for the star trackers.

4.3.3.5 System Specifications. Estimates of weight, power, and volume for the subsystem components are shown in Table 4-6. Power consumption for each satellite is totaled in Table 4-7.

Table 4-6. Component Weight-Power-Volume Estimates

COMPONENT	WEIGHT (lb)	VOLUME (in.^{3})	POWER (watts)
Star Tracker	15	1000	13
Horizon Seeker	15	300	13
Computer	35	1000	30
GIMU Package (Strapdown Gyros)	10	300	30
Logic Unit (Data and Ranging)	20	500	15
Others (Contingency, Signal Conditioners, Buffers, etc.)	20	500	10

Table 4-7. Navigation and Attitude System Specifications

A. STAR TRACKER (Kollsman Solid State Tracker)

General
- (a) Solid state electronics
- (b) Two-position gimbals, torque motors respond to digital command from external computer
- (c) Point telescope within 120° cone angle

System Size	Tracker	Electronics
Volume	< 1000 in.^{3}	< 100 in.^{3}
Weight	< 10 lb	< 5 lb

Power   < 15 watts   (13 watts maximum)

Reliability   Designed for one-year operation in space

Performance

Accuracy   15 arc sec per axis (2σ) (5 Arc minutes required)

Recognition   Acquire and track stars of +1.7 silicon magnitude and brighter

Acquisition Rate   0.5 deg/sec (Acquire star for apparent velocity)

Sun/Earth Impingement   Unimpaired accuracy: as close 25° to sun; 1° to earth

Table 4-7.   Navigation and Attitude System Specifications, Contd

<u>Environmental</u>

      Temperature $0°$F to $150°$F

      Acceleration   11.3 g (all axes) duration
                                       4.5 minutes/axis

B.   <u>HORIZON SEEKER</u>   (ATL Advanced Sensor)

<u>General</u>   (a)   Edge Tracking Technique
                            (b)   Insensitive to IR Radiation Level
                            (c)   No Bearings
                            (d)   High Altitude Application

<u>Size</u> < 300 in.^{3} (total includes electronics)

<u>Weight</u> < 15 lb

<u>Power</u> < 15 watts (13 watts maximum)

<u>Reliability</u>   Designed for one-year operation

<u>Performance</u>

      Accuracy $\pm \leq 0.1°$ per axis roll/pitch

      Scan Range   Elevation $\leq 85°$
                                 Azimuth $\leq 120°$

      Linear Range $\geq 10°$ pitch/roll

      Time Constant ≤ 100 ms

      Altitude Range $100 \rightarrow 80,000$ n.mi.

<u>Environmental</u>

      Temperature $-30°$F to $165°$F

C.   <u>DIGITAL COMPUTER</u>   (Preliminary, similar to IBM 4π)

<u>General</u>   (a)   General purpose, stored program, binary operated
                                        airborne computer
                            (b)   Memory: random access, parallel readout
                            (c)   Parallel processing
                            (d)   Redundancy: parts redundancy
                            (e)   Arithmetic: parallel processing

<u>Size</u>

      Volume ≤ 1000 in.^{3}

      Weight ≤ 35 lb

Table 4-7. Navigation and Attitude System Specifications, Contd

Power	≤ 30 watts (see notes)
Reliability	≥ 5,000 hr, MTBF (see Note 2)

Performance

Word Length: (May be decreased — see Note 1)	(a) Instruction: 32-bit instruction words (b) Data: 32-bit data word	
Instruction Times: (For the speed-up with current microcircuits)	(a) Add/Subtract < 5.0 μsec (b) Multiply < 10 μsec (c) Divide < 15 μsec	
Memory Access Time	-1.0 μsec	

Notes:

1. Computer specifications covering speed, capacity, word length, and power should be finalized only after total computational tasks have been defined; also required accuracy and operating periods.
2. It is assumed that during some periods, we can shut off the computer.
3. Power indicated is an average estimate made by evaluating Saturn, Apollo, IBM 4π and micro-electronic 1824 computers, and forecasting for 1970.

D. STRAPDOWN ATTITUDE UNIT   (GIMU for Lower Satellite)

General	(a) All altitude (3 gyros) (b) Digital angular pickoff on pulse torque (c) Single package includes associated electronics for torquing, heaters, etc.

Size

Volume	< 300 in.^{3}
Weight	< 10 lb

Power	≤ 30 watts
Reliability	Designed for one-year operation (using gas-bearing gyros)
Performance	(3) gyros - gas-bearing (Typical MH GG 3346) Gyro drift (total) < 1.5 deg/hr

Table 4-7. Navigation and Attitude System Specifications, Contd

E. RANGING AND DATA LINK, LOGIC UNIT   (Estimate)

Size	< 500 in.^{3}
Weight	< 20 lb
Power	< 15 watts

F. OTHERS: Signal Conditioners, Buffers, Contingency, etc.

Size	< 500 in.^{3}
Weight	< 20 lb
Power	< 10 watts

The upper satellite power consumption could be reduced by removing the upper computer and performing all calculations by time-sharing the lower satellite computer. The configuration is shown in Figure 4-23.

The disadvantages are:

a. Increase in lower satellite computer loading and capacity.

b. Increase in lower satellite computer power consumption.

c. Increase in data link capacity.

The data in Table 4-6 yield 115 lb of equipment for each satellite.

4.3.3.6 Component Specifications. Comprehensive specifications for the components summarized in Table 4-6 are given in Table 4-7 for typical equipment. A brief general description is given for each component. Weight, volume, power consumption, reliability, performance, and environmental data are included.

4.3.4 Power. The power subsystem is a conventional system as indicated in the following diagram. However, it represents an integration of such specific factors as power requirements, source and storage provisions pertaining to this experiment.

In this experiment, the operational mode with the most severe power requirement happens to be the mapping mode. This is also the mode that consumes more than 90% of the total elapsed time of the nominal mission. Consequently, the power system is tailored to this condition with allowance for exceptional situations.
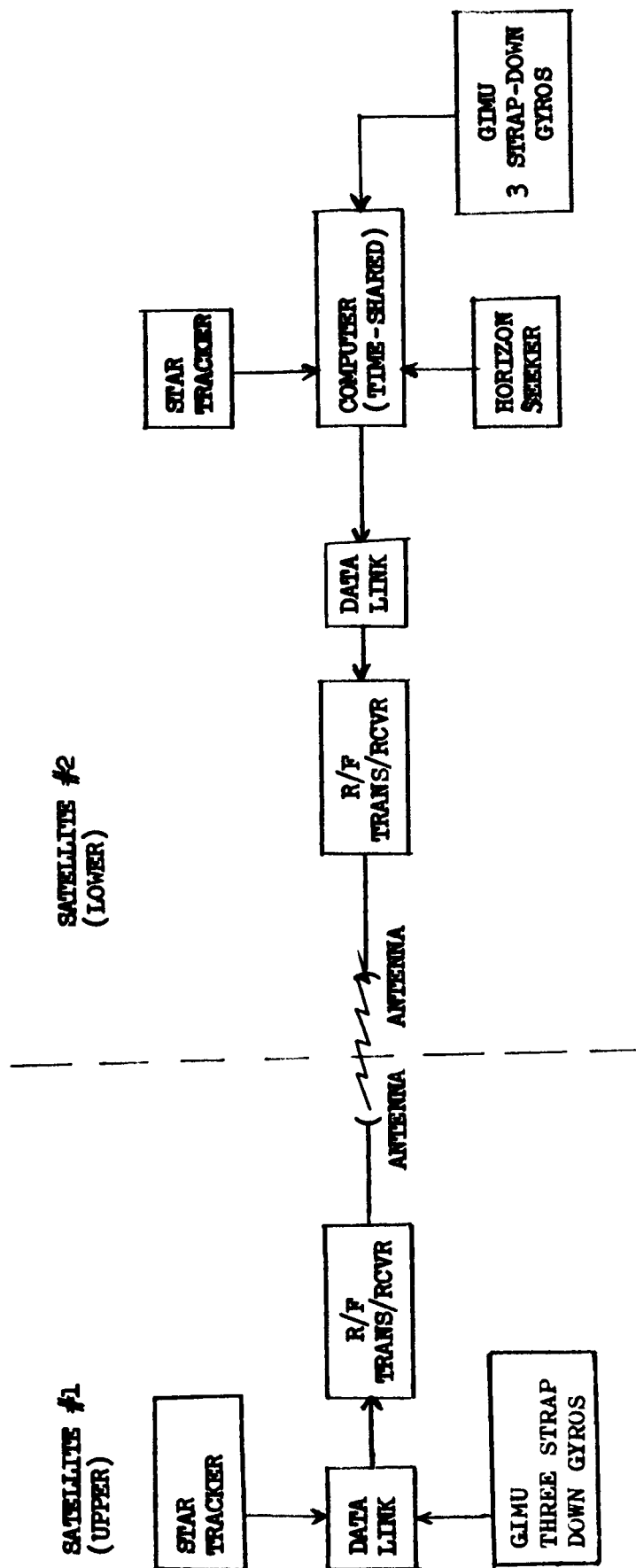
Figure 4-23. Possible Minimum Power Configuration Using One Time-Shared Computer

```
              SOLAR
              CELLS
                │
                ▼
   ┌────────────┼───────────BATTERIES
   │            │                ▲
   │            ▼          ┌──────┘
   │          POWER        │
   │        CONDITIONING   │
   │            │
   ▼            ▼
POWER DISTRIBUTION
```
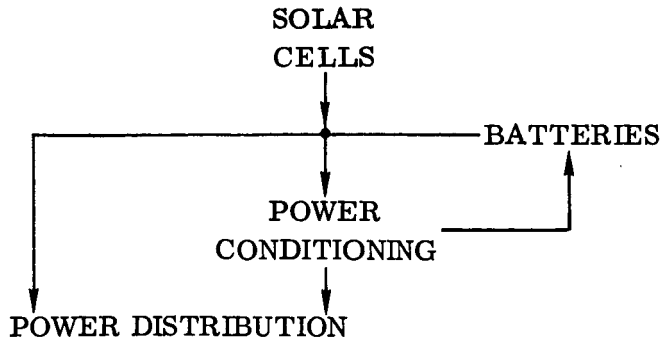
The source of power is a rather obvious choice. Cost rules out radioisotope thermo-electric generators. Fuel cells require a large volume, which is at a premium. The remaining choice is between different types of solar cells. The cadmium-sulfide cell at 5.5 watts/ft^2 is discarded in favor of the silicon cell at an estimated 13 watts/ft^2 for the time period involved. The weight penalty of the one selected is less significant than the required area penalty of the other.

With a calculated effective solar cell area of 115 ft^2, the power available is an average of 307 watts. This provides a slight margin over the maximum average required power and this margin is augmented by the periods of less than average consumption.

The power storage requirement is based on an assumed drain of twice the power requirement for 1.2 hours dark time. This requirement of 723 watt-hours is met with 27 silver cadmium batteries connected in series to provide 28 volts dc. Assuming 83 ampere hours capacity and a 40% drain, these cells can provide 900 watt-hours.

Recharge is accomplished with an 80% efficiency in 6.3 days during the maximum average consumption mode.

Power conditioning equipment is required to convert the output of the solar cells into the voltages, currents, and frequencies required by the antenna equipment. Dc regulators and dc/ac inverters for space application are available in the 1.0 kW range. Based on current BSM designs, such a unit, including pulse-width-modulated regulator, should not exceed 15 lb.

Operating voltages and frequencies are dependent upon optimum interfacing between power source and loads. Standard 28 volt dc and 115 volt 400 Hz will be used except when special considerations demand exception. Three phase power will generally be 115/199 volts ac.

Consideration of higher voltages will be reviewed in the light of possible weight savings and other advantages, but it is anticipated the advantages of cost, availability and relia-bility will confirm the use of the standard power parameters noted.

Protection devices to guard against overload or open circuit are incorporated together with redundant components and circuitry as necessary to obtain the reliability required for the length of the mission.

4.4 DYNAMICS AND ATTITUDE CONTROL. The major feasibility problem inherent in the dynamics and attitude control technology area for the crossed-H concept as envisioned in this study lies in the dynamic behavior of the tether and the two satellites, particularly during the slow separation change operation. It is believed that the presence of active jet engines on both satellites and computer(s) will allow control of the structure to the desired dynamic behavior. This assumption will have to be the subject of a detailed simulation during future work.

Docking dynamics have not been examined. Since the individual satellite is small compared to the CSM, the docking impulse imparted to the satellite, particularly for a non-latch contact, may cause appreciable motion. This problem, however, is minimized by the retraction of the booms and dipoles during docking.

Dynamic behavior during operation will be satisfactory. The dynamic behavior of structure extending out from satellites has caused problems in the past. For example, the large solar panel array originally configured for MORL introduced undesirable coupling into the control system. The crossed-H dipoles are the primary concern here. Estimates of their dynamic characteristics show that the dipole oscillations will be within acceptable limits. Boom oscillations should be smaller.

There are no other feasibility questions in the dynamics and attitude control task. The remainder of the work done was examining predesign considerations. For example, a rigid body can be controlled to any attitude which is sensed; the predesign engineer must select actuators, sensors, examine weight, reliability, and cost. The studies are detailed in the subsequent discussions.

4.4.1 Payload Separation. The use of the CSM to remove the payload from the experiment provides positive manual control over this operation. The experiment is oriented to the local vertical by the CSM. The satellite attitude control system then separates the satellites and extends the tether along the vertical. Since the jet engines are low thrust cold gas, one satellite's jets will not damage the other satellite during the initial phases of extension. The extension is observed by the astronaut, who may use his override capability at any time. This procedure is simple, direct, and quite reliable.

The separation distance will be varied automatically during the mission. However, the two satellites will not be brought back together except under man's supervision.

4.4.2 Complete Experiment Dynamics. The complete experiment consists of the two satellites connected by the mylar tether. Nominal alignment is with local vertical. Gravity gradient will hold the experiment close to this orientation. Satellite separation

can be varied by extension and retraction of the tether. Active attitude control is provided for each satellite.

As the configuration moves along the orbit, the experiment can scan or observe specific sources. During scan, the satellites maintain a specific orientation with respect to the local-vertical/orbital-plane frame. During source observation the satellites are rotated to point at a source, held for the desired observation time, and then rotated to the next source. The use of active attitude control makes possible source observation and scan positions other than the gravity gradient stable orientation. Designs that do not control the rf antenna direction (i.e., satellite orientation) do not have these capabilities.

The presence of the tether provides the following advantages. Separation is controlled by positive mechanical methods, and the environmental gravity gradient phenomena is utilized as the basic control torque for the entire structure. The tether provides backup for subsystem failure, maintaining the satellites in proximity until EVA activity corrects the failure.

Vertical Orientation. The experiment is nominally aligned with local vertical to use gravity gradient for control of the major motions. The satellites are held at the desired separation distance by the tether.

Tether integrity has been examined in detail by Johns Hopkins in four areas. These are:

a. Severing by meteoroids.

b. Degradation in space environment.

c. Excessive bowing due to solar radiation.

d. Excessive twisting due to relative rotation of the two end bodies.

These problems are satisfactorily resolved by Johns Hopkins. Items a, b, and c are directly applicable to this crossed-H concept, and so were not investigated further.

The Johns Hopkins method of preventing twisting uses magnets. The concept under discussion here has arbitrary orientations with respect to the earth's magnetic field. The magnet method would produce disturbance torques and is not acceptable. Twisting is prevented by the active attitude control of each satellite, plus utilization of a solar pressure gradient.

The dynamic characteristics of the tether can be examined by considering the structure to be two masses connected by a spring. The frequency and spring constant are shown in Figure 4-24 as a function of tether length for a mylar G tape 1.5 mil thick and 0.75 in. wide.
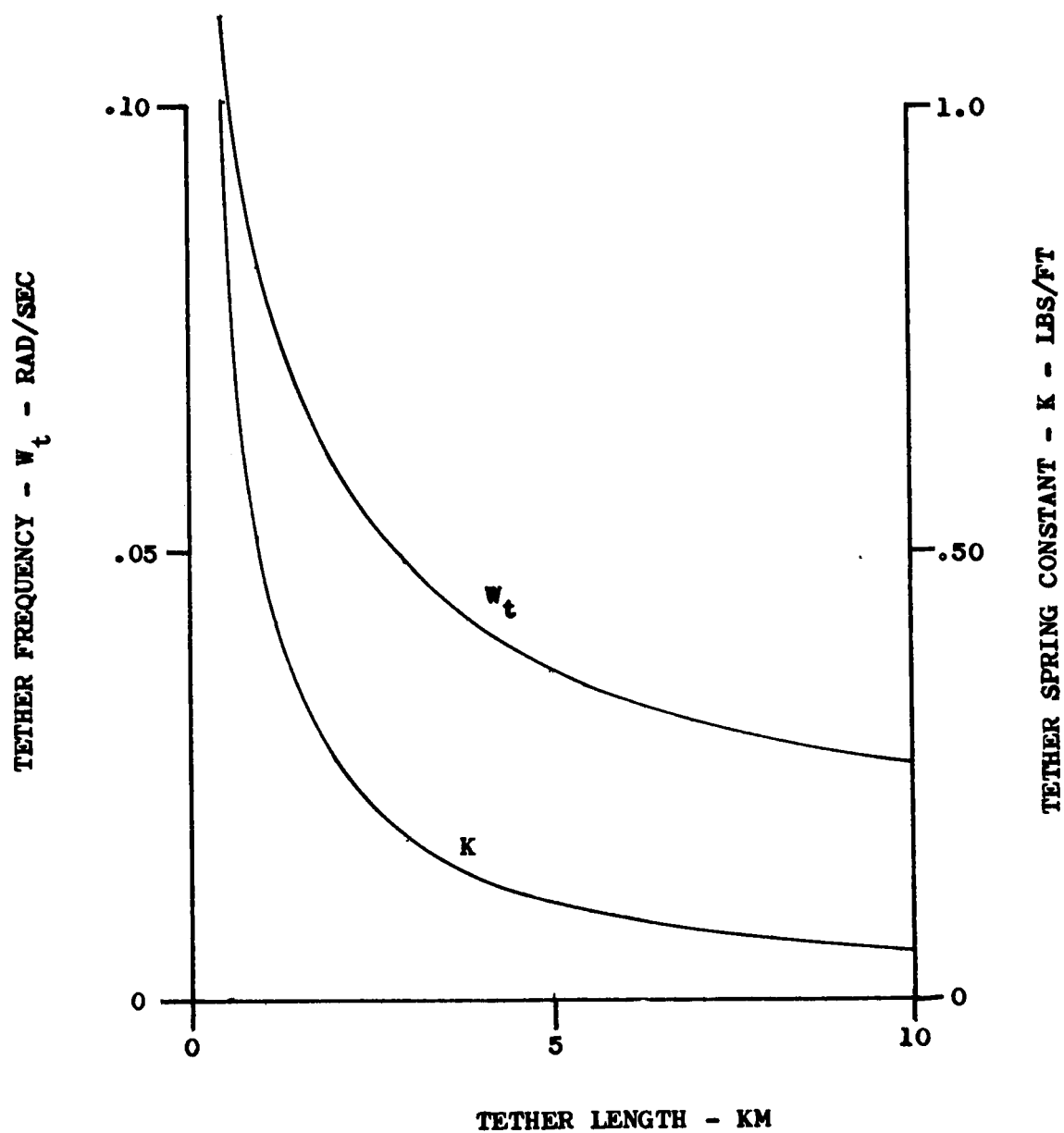
Figure 4-24. Tether Dynamic Characteristics vs. Length

Tether frequency at longer lengths is close to the dipole frequency at longer lengths, as shown in Figure 4-29. Consequently the larger configurations will probably have some tether-dipole coupling. The dynamics data are used to estimate the tape tension. When there are no perturbing dynamic motions, the complete structure is controlled by gravity gradient. This steady state tape tension is then proportional to length, being 0.02 lb at 10 km. In practice, dynamic effects such as tether-dipole coupling, jet firings, tether oscillations, and end body motions, will usually be present, and higher tensions result.

Maximum tension is estimated by calculating the effects of firing two engines for 10 seconds and allowing the tether to arrest the motion. Impulse is 2 lb-sec, and end body velocity is 0.0257 ft/sec. Resulting parameters are:

Tether Length (km)	Tether Stretch (ft)	Maximum Force (lb)
10	0.97	0.053
0.5	0.22	0.24

Tension is considerably above the steady state value, but considerably below the breaking point. The tether is protected by the tension take-up reel.

The motion of a similar system has been investigated in Reference 6. It is pointed out that although the environment consists of only conservative force fields, instabilities can exist in the sense that small disturbances can grow with time. A more detailed examination of the tether/end-body dynamics is necessary.

Gravity Gradient. The major motions of the complete experiment are with respect to the local-vertical/orbital-plane under the influence of gravity gradient. The period of these oscillations is of the order of 12 hours. Oscillations are excited by several sources, orbit eccentricity being a major one.

A damping mechanism must be provided. The work by Johns Hopkins shows that viscous damping dependent upon tether-satellite motion is not in general satisfactory for this type of structure. It is not desirable for this particular design to anchor to the earth's magnetic field through the use of magnetic viscous damping. Since jet control is required to be aboard for other purposes, it is convenient to damp the libration motions with jet control.

Extension and Retraction. Separation is changed by extending or retracting the tether. There are two different ranges of separation rate, one is a slow change of the orders of tens of meters per orbit during which rf measurements are being recorded. The other is a faster change of the order of several thousand meters per orbit in which no measurements are taken. The faster changes are done solely to change tether length, while the slower changes form a major portion of the experiments.

4-65

The angular momentum of the complete experiment must be changed as the tether length changes. The experiment rotates at earth rate to remain aligned with local vertical, and the jets introduce appropriate angular momentum increments as the length changes.

The net force due to gravity and centripetal reaction acts to keep the satellites separated and the tether taut. This aids tether extension and opposes tether retraction.

The tether change will be initiated by imparting a jet impulse to each satellite to provide the desired separation velocity. The reels will take up or pay out the tether at this velocity. Thus primary control of the average velocity is provided by the reels. The jets will be used for the additional control functions necessary as the satellites move. These include changing angular momentum, damping libration motions, preventing tape oscillation buildup, and precluding any detrimental motions from the various possible couplings of tether and satellites.

Note that adequate control of these changes requires accurate information on the separation and separation rate. This cannot be obtained by sensing tether motion as it enters the reel. The separation data will be supplied by a radar.

The feasibility of the tether operations is the most significant problem in the dynamics and attitude control area. Operation is more complex than for the design investigated by Johns Hopkins because of the tether length changes and the slow changes during which measurements are made. The attitude tolerances on motion are more stringent. Thus a more complicated dynamics behavior will occur.

The level of this study precludes detailed dynamic studies such as that by Johns Hopkins. Feasibility is indicated by the following considerations. Since the vehicle is at synchronous altitude, the various perturbations, such as orbit eccentricity and solar pressure, are at low frequencies and are small. Pertinent structural frequencies, such as dipole and tether first modes, are also low. Therefore the dynamic behavior will be slowly varying phenomena involving small forces. The presence of the jet thrusters and an autopilot computer provides the capability of correcting undesirable behavior in time periods small compared to the periods of the dynamics. Thus it should be possible to control the tether and the two satellites to the desired operational movements.

It is necessary that these considerations be investigated by a complete dynamic analysis. This will require the development of an extensive simulation, and a large number of runs will be needed to investigate the operation performance. The more pertinent effects that may be included in the simulation are:

a. Environment torques, gravity gradient, and solar.

b. Orbit eccentricity.

c. Libration motions.

d. Jet engine damping of libration motions.

e. Spring effect of tether.

f. Spring effect of gravity and centripetal net force.

g. Change of angular momentum due to tether length change.

h. Coupling between tether spring oscillation and dipole first mode.

i. Coupling between tether spring and satellites due to:

    1. Tether tension acts through cg offset.

    2. Change in tether tension due to jet engine firings.

    3. Limit cycling.

j. Accuracy of measurement of separation and separation rate.

k. Perturbations introduced by dipole and boom length variations.

l. Reel angular momentum.

4.4.3 <u>Individual Satellite</u>. Each of the two satellites have nearly identical attitude control systems. These systems maintain the desired orientation and, acting together, control the motion of the complete experiment. The attitude control system is discussed in Section 4.4.3.1. The pertinent structural dynamics considerations of the individual satellites are examined in Section 4.4.3.2.

The weight of the selected jet system is estimated to be 282 lb for each satellite. The sensing instrumentation and computer subsystem equipment for both navigation and attitude control is 397 lb per satellite.

<u>Actuation Devices</u>. The impulse requirements that the actuators must meet are due to:

a. Tether extension and retraction.

b. Angular momentum changes during tether extension and retraction.

c. Libration damping.

d. Environmental disturbances.

e. Maneuvering.

f. Limit cycles.

g. Corrections for various system perturbations such as satellite-tether coupling, engine misalignment, and "tether spring" oscillations.

These impulse requirements demand six-degree-of-freedom control of each satellite, three of rotation and three of translation. The actuators were selected to be jets, as a result of the subsequent discussion. Roll control and two of the translation forces will be supplied by jets mounted on the center of each satellite, pitch and yaw control and the third translation force will be supplied by jets mounted on the dipole heads at the boom ends (see Figure 4-25).

There are four sets of jets, each set containing four jets. The two sets on the satellite center are supplied by a single tank. There is a set of jets and a tank on each of the two dipole heads. Failure of any of the 16 jets will degrade the mission. A portion of the jets have functions that can be supplied by another jet. This will result in the propellant being expended from a particular tank sooner than planned. Note that the telescoping boom prevents transfer between the tanks. Another portion of the jets have functions that cannot be supplied by others. If one of these fails, the experiment cannot continue until repairs are accomplished. Therefore, jet reliability is an important consideration for mission success.

Since the individual satellite requires active attitude control to remain in the local horizontal plane, ACS failure will result in a vertical orientation. In this eventuality the dipoles and booms will be fully retracted to avoid tether contact. The complete experiment will be passively controlled by the tether to local vertical by gravity gradient until repairs occur.

The impulse and weight requirements for the full mission are shown in Table 4-8. The data would be 78% of that shown for resupply at the end of one year.

The data for libration damping and corrections are approximate estimates. They must be calculated more accurately during tether dynamic analyses. Since these approximate estimates form the major portion of the requirements, the resulting ACS weights are subject to future refinements.

The propellant choice considered N_2 and H_2O_2. The nitrogen cold gas is more reliable than H_2O_2, but weighs more and occupies a large volume. N_2 is selected as the propellant to obtain higher reliability with acceptable associated system penalties. The weight of N_2 does not appear to be a problem. The tank volume required for the satellite center body may be a problem. A part of the center body jet requirements can be shifted to the dipole heads. The selection of N_2 also allows a smaller minimum impulses. This is important in keeping limit cycle requirements low and making subtle corrections to separation velocity. The thrust level for all jets is 0.1 lb. Data in Table 4-8 for limit cycles assumes this value.

The actuator selection is based on the following considerations. Both translation and rotation control must be available. The translation requires engines, such as jets or electric propulsion. The occurrence of cyclic phenomena in the requirements for rotational torque allows utilization of momentum storage devices. This has the potential
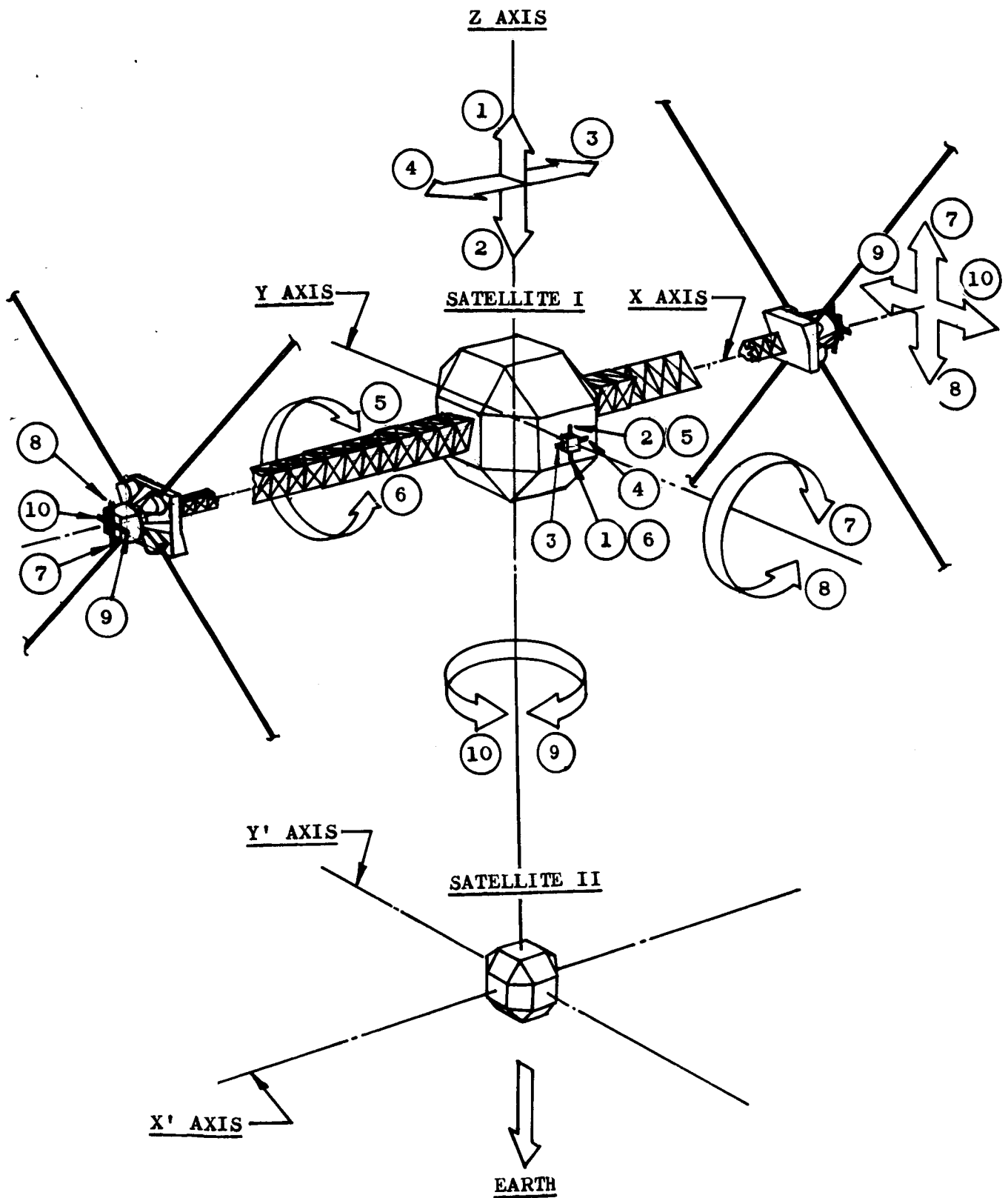
Figure 4-25. Geometric Reference Axes and Attitude Control Geometry

Table 4-8. Individual Satellite Actuator Impulse Requirements and ACS Weight

	SATELLITE CENTER	TOTAL FOR TWO DIPOLE HEADS	TOTAL FOR SATELLITE
	Impulse Requirement (lb-sec)		
Tether Motion	52		
Angular Momentum	328	978	
Libration Damping	723	723	
Environment	6	58	
Maneuvering	-	139	
Limit Cycles	49	64	
Corrections	4130	1230	
Totals	5288	3192	8480
	Weight of ACS Due to Tanks, Valves, Nozzles, and Propellant (lb)		
N_2	176	106	282
H_2O_2	38.9	23.4	62.3
	Propellant Weight (lb)		
N_2	88	53	142
H_2O_2	33.1	20	53.1

advantages of reducing ACS weight due to lower fuel consumption and improving engine reliability by reducing firing frequency. The utilization of momentum devices introduces additional complexities, and requires resetting (desaturation).

Control system actuator selection is subject to tradeoffs involving:

a. Accuracy.

b. Performance.

c. Dynamic response.

d. Energy expenditure.

e. Complexity.

f. Weight.

g. Reliability.

Due to a long operational period reliability may be the most important factor. The systems considered were:

a. Inertia wheels and jets.

b. Fluid flywheels and jets.

c. Control moment gyros and jets.

d. Mass expulsion (jets).

e. Electric propulsion.

Table 4-9 summarizes the advantages and disadvantages of each system.

The selection criteria include near-future application. Further, desired momentum storage capacity is low. This allows two choices, mass expulsion (jets) or the inertia wheels and jets combination.

The ultimate decision between these two will be made in future tradeoff analyses. However, preliminary considerations favor the all-jet approach and it is selected for the predesign system.

The potential weight savings of the combination are low because only a small portion of the total impulse requirements are cyclic. The effects on reliability are not immediately obvious. The N_2 propellant selection and the redundant valves constitute an extremely reliable jet system. Pending detailed tether dynamics analysis, the potential reductions in jet firings are not firmly established.

Sensors. The attitude control function requires the following information.

a. Attitude and attitude rate of each satellite.

b. Relative attitude and attitude rate.

c. Separation and separation rate.

The attitude control system must control the satellite rigid body motion to ±0.1 degree in order to meet the system requirements in the presence of dynamics and thermal deflections. Consequently, at least one star tracker is required. During normal operation, the satellite rate information will be obtained by electronic circuitry, for example,

Table 4-9. Attitude Control Actuator Tradeoff

	ADVANTAGES	DISADVANTAGES
I INERTIA WHEELS AND JETS	1. RELATIVELY SIMPLE 2. ADVANCE STAGE OF DEVELOPMENT 3. LOW POWER REQUIREMENTS 4. HIGH ACCURACY CONTROL	1. HIGH PEAK POWER REQUIREMENTS 2. GYROSCOPIC CROSS-COUPLING 3. LIFE LIMITED DUE TO MOVING PARTS
II FLUID FLYWHEELS AND JETS	1. SIMPLE MECHANICAL RANGE 2. HIGH RELIABILITY (NO MOVING PARTS) 3. RAPID RESPONSE 4. HIGH/LOW TORQUE CAPABILITY	1. NO WEIGHT AND POWER SAVING COMPARED TO I 2. IN STATE OF DEVELOPMENT 3. METHOD OF RESET STILL PROBLEM
III CONTROL MOMENT AND JETS	1. RAPID, LOW POWER, HIGH SPEED HIGH TORQUE RESPONSE 2. INHERENTLY STABLE 3. STABILIZATION CAN BE OPEN LOOP WITHOUT ELECTRICAL POWER	1. HIGHLY COMPLEX 2. LIFE LIMITED DUE TO MOVING PARTS
IV MASS EXPULSION (JETS)	1. SIMPLE 2. ADVANCED STATE-OF-THE-ART 3. LOW POWER REQUIREMENTS	1. SYSTEM LIMITED BY WEIGHT(FUEL) 2. LARGE EXPENDITURE OF FUEL 3. POSSIBLE CONTAMINATION OF SURROUNDING DUE TO FUEL EXPULSION
V ELECTRIC PROPULSION	1. ACCURATE CONTROL OF TORQUE 2. HIGH SPECIFIC IMPULSE 3. SUITABLE FOR LONG MISSIONS	1. LOW TORQUE 2. HIGH POWER REQUIREMENTS

by psuedo rate circuits or digital computer calculations. The relative attitude motion of the two satellites will be calculated by computer from the attitude data.

Accuracy requirements on the separation and separation rate information will be obtained from the tether dynamics study.

The sensor instrumentation and digital computing requirements for the attitude control functions are combined with the navigation function requirements, resulting in an integrated navigation-attitude control equipment subsystem. This integration and the resulting hardware components were discussed in Section 4.3.4.

4.4.3.2 Structural Dynamics. The only portions of the individual satellites that are of concern are the dipoles and the booms.

Booms. The booms are shorter and stiffer than the dipole, and are subjected to approximately the same loading. Since the dipole is acceptable, it is not necessary at this point in time to confirm that the boom is acceptable. It will be necessary to check the boom response during the design phase.

Dynamic Characteristics of the Dipoles. The two most important characteristics are tip deflection due to disturbances and first mode frequency. These characteristics are estimated here. All dipoles are identical, being uniform and containing no tip masses. For any of the several experimental configurations, all dipoles have the same length. Hence, the dipole dynamics are obtained by considering a uniform, cantilevered beam representation.

Major dipole tip deflections are due to jet firings. They will be most detrimental to the rf experiment at maximum dipole length, and data are presented for this situation. The deflection can be reduced by increasing dipole diameter; unfortunately, increasing the diameter increases thermal distortion as well as the lesser penalty of increased weight. Consequently, the dipole diameter selected for design results from tradeoff studies of the dynamics and the thermal distortions. Dynamic deflection data are therefore presented as a function of dipole diameter.

The jet firing disturbances to the dipoles are of two types, couples to produce rotation and forces to produce translation. The couples are applied for short time periods relative to the dipole frequency, and thus are effectively an impulse disturbance. The translation forces are produced by jet firings, which are long in time relative to dipole frequency (e.g., libration damping, tether extension), and thus are effectively represented by a steady state force.

Deflections are shown in Figure 4-26 for rotational impulse and in Figure 4-27 for translation force. The deflection in roll decreases more rapidly than in pitch or yaw with dipole diameter because the larger dipole has a greater relative effect on the roll inertia. The largest anticipated rotation impulse would be for desaturation of momentum
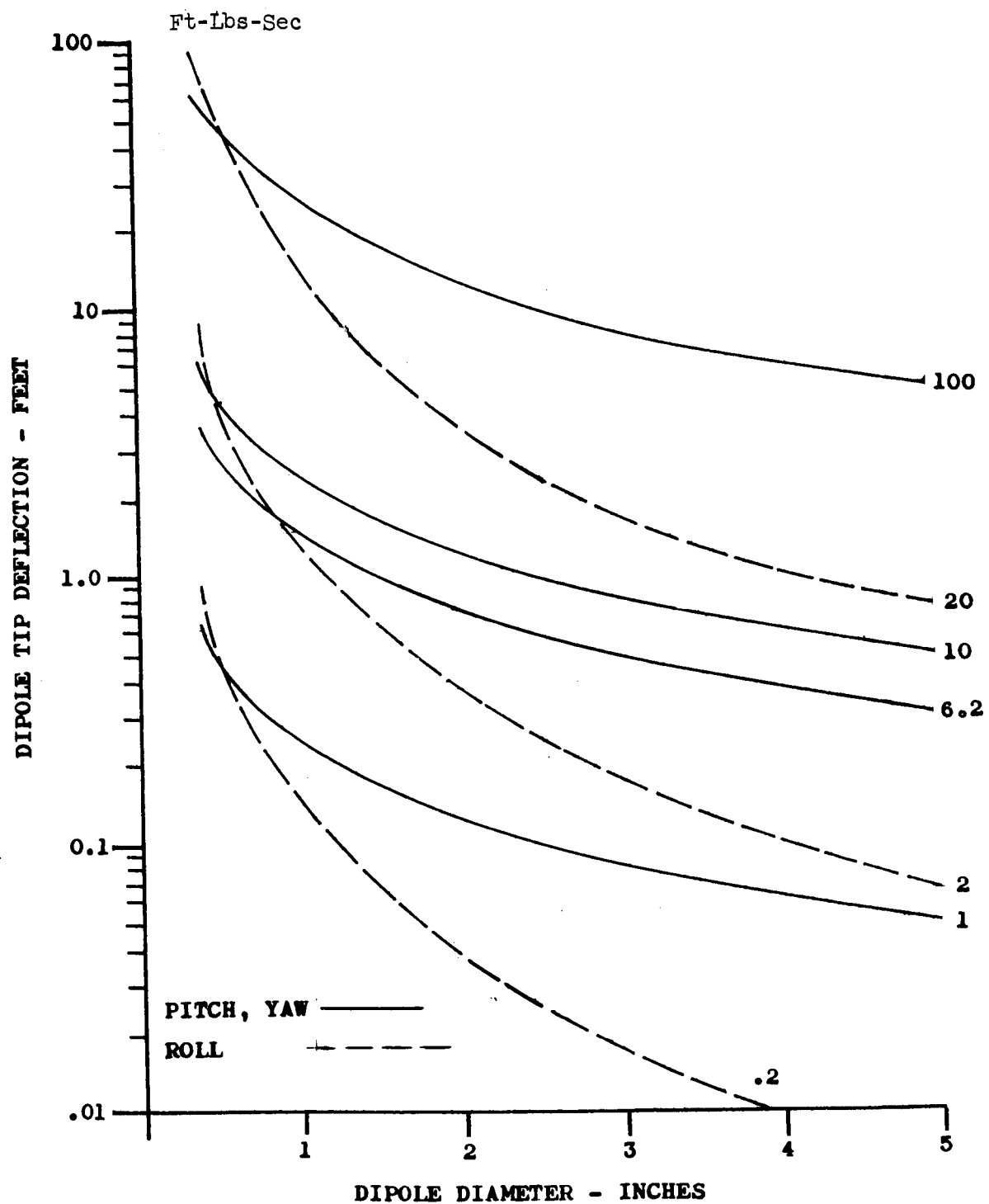
Figure 4-26. Deflection of Fully Extended Dipole Element vs.
Ft-Lb-Sec of Rotational Thrust Impulse
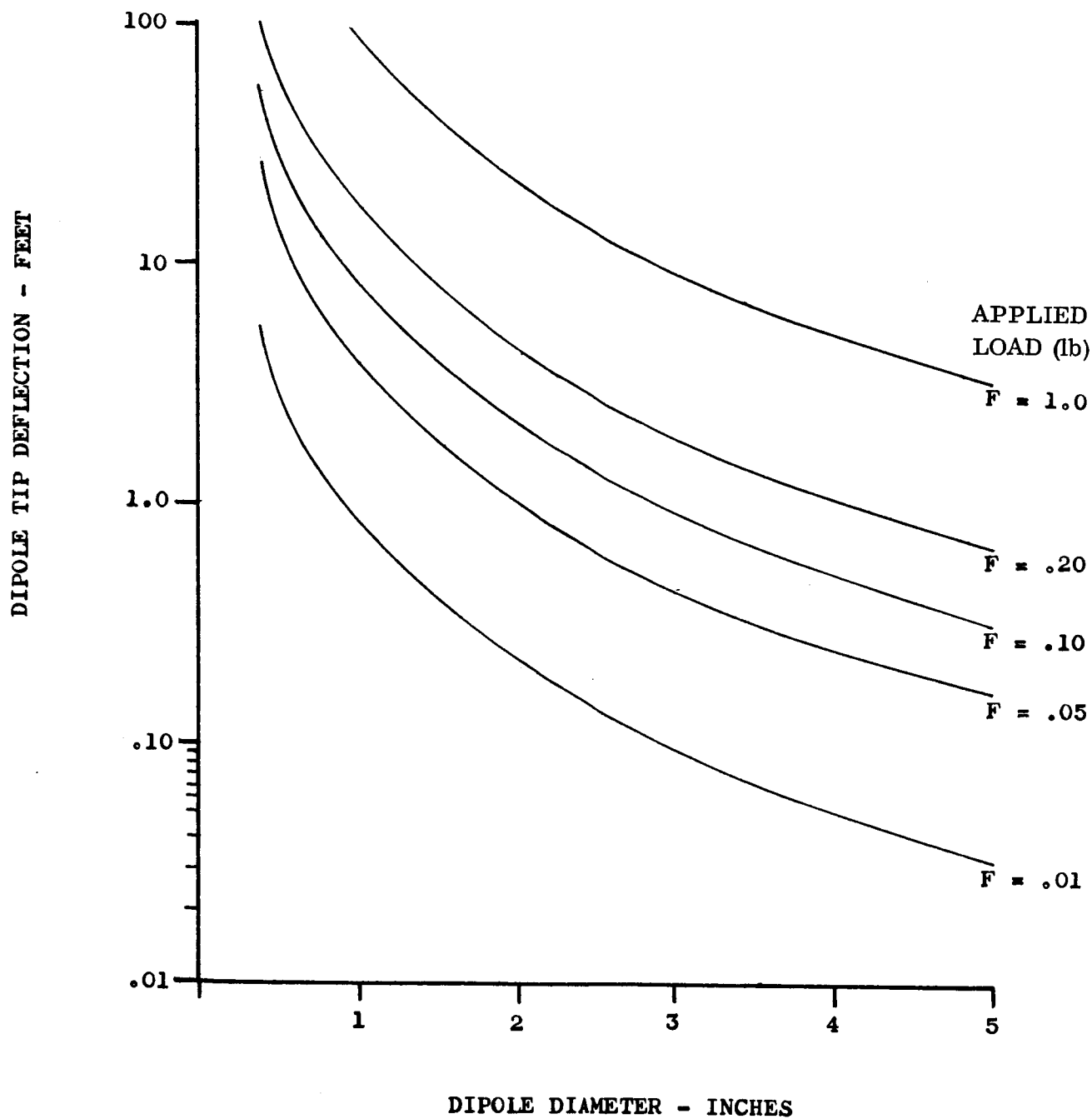
Figure 4-27. Dipole Deflection vs. Applied Load

devices, should they be included in the design. Currently, they are not. Such devices might be of 10 ft-lb-sec in pitch and yaw and 2 ft-lb-sec in roll. The next major rotational disturbance is maneuvering from target to target. Maneuvering at a velocity of one radian per hour requires impulses of 6.2 ft-lb-sec in pitch and yaw. No other rotational disturbances of this magnitude are anticipated during normal operation. Dipole deflections due to rotational disturbances will not be larger than approximately 9 ft for diameters of an inch or greater. These deflections are not detrimental.

The design uses engines of 0.1 lb, and they are fired in pairs for translation. Thus the applied force will be 0.2 lb. Recall that the data in Figure 4-27 is for an applied force time period greater than the dipole period. If the engines are on for a shorter time period, the deflection will be less. A maximum tip deflection of 16.5 ft will occur with 1 in. dipole diameter. The deflection varies rapidly with diameter.

First mode frequency is of interest for two reasons. First, it indicates the frequency area in which coupling with other structural or control frequencies can occur. Second, from the dynamics point of view, the first mode frequency is a measure of the structural stiffness. It is convenient from this point of view to have the first mode frequency. The frequency for the maximum dipole half length, is shown in Figure 4-28 as a function of dipole diameter. The frequency for the 1-in. diameter is 0.0231 radians/sec.

The frequency for the 1-in.-dia. dipole is shown in Figure 4-29 as a function of dipole length. For the shorter lengths, the frequency rapidly increases.

Once a dipole is excited into oscillation, the amplitude will decrease with time depending on its inherent damping. The dipole mesh is welded at each wire-to-wire contact, and the effective damping ratio is about 1%. The amplitude will decay to one half in 10 cycles, which is 50 minutes.

4.5 ANTENNA PERFORMANCE

4.5.1 Validate Choice of Variable Geometry Crossed-H Interferometer. The primary long wave radio astronomy (LWRA) objectives are discussed in Section 2.3; scientific requirements for instrumentation to meet these objectives are discussed in Section 3. In summary, the principal measurements desired of a LWRA system are 1) the spectral brightness and polarization mapping of essentially time-stationary sources for frequencies below 10 MHz and 2) the spectral brightness and polarization monitoring of strong time-varying sources within the solar system for frequencies below 10 MHz.

Earth-based radio telescopes are limited in their usefulness in varying degrees below roughly 30 MHz by the reflection, absorption, refraction and polarization rotation effects of the ionosphere. They are also adversely affected by interference from manmade signals and atmospheric noises originating on the earth. These limitations increase in severity with decreasing frequency, becoming very severe at about 10 MHz
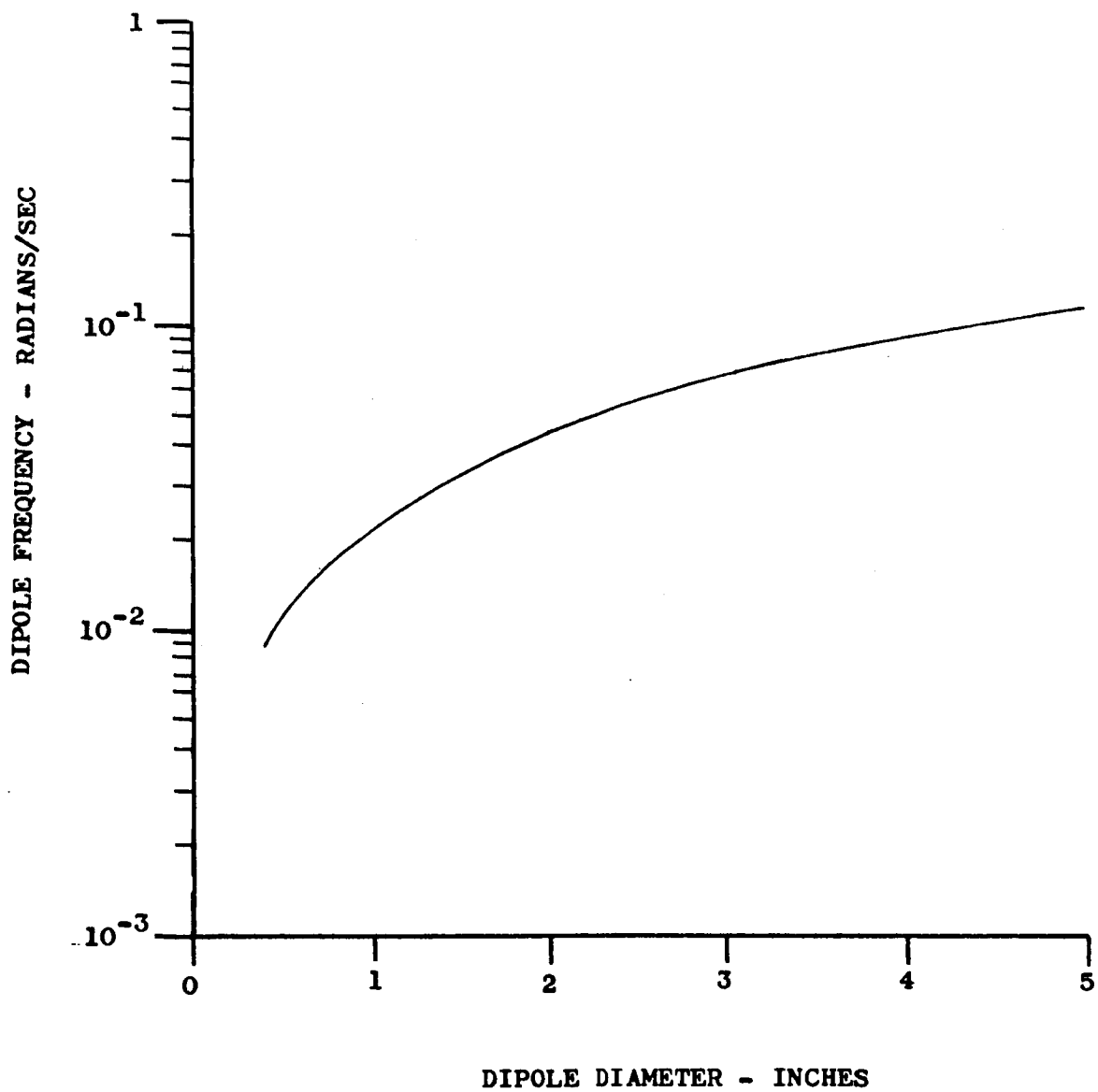
Figure 4-28. First Mode Dipole Frequency vs. Dipole Diameter
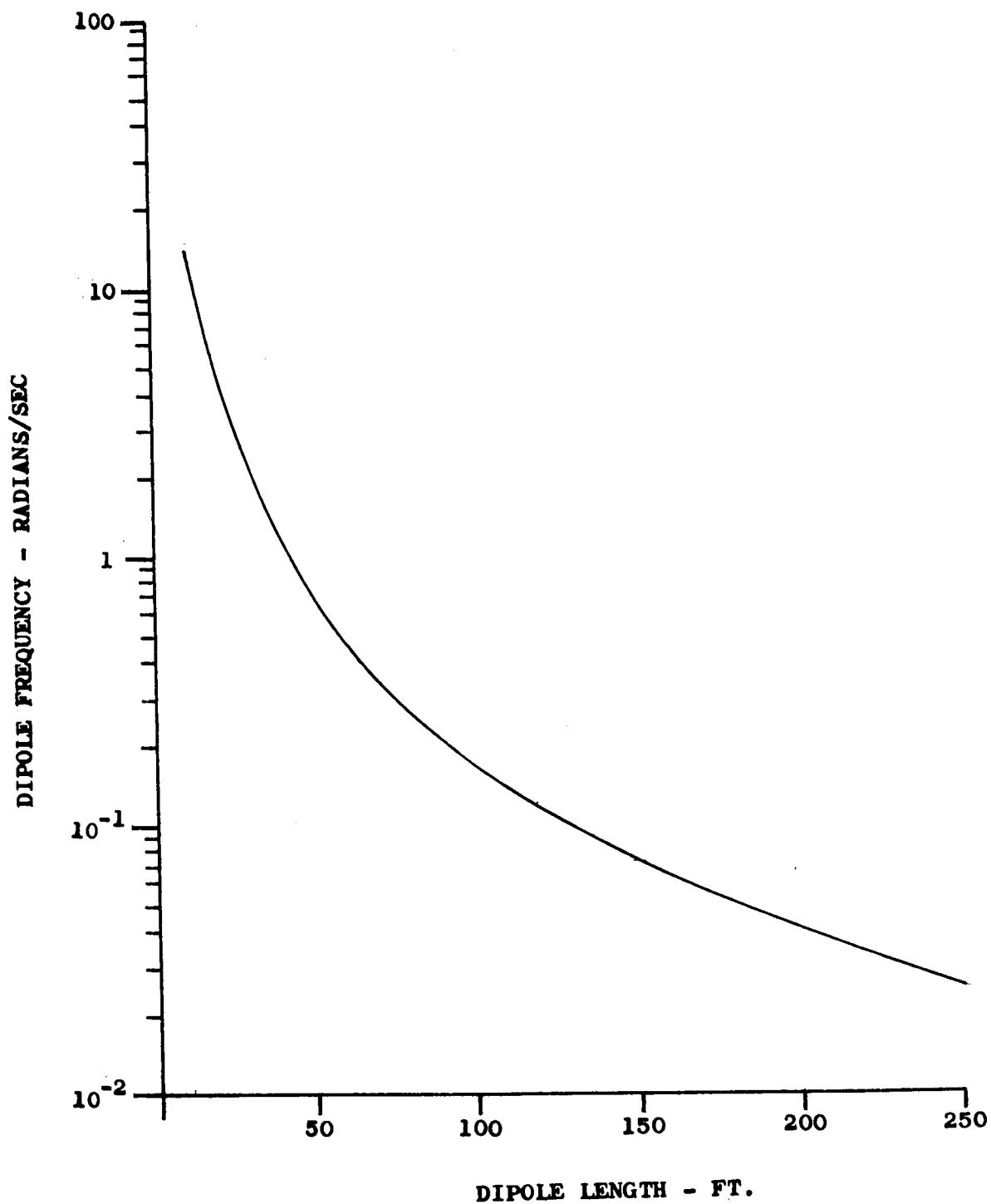
Figure 4-29. First-Mode Dipole Frequency vs. Length
for 1-In.-Dia. Dipole

and intolerable at frequencies below about 5 MHz. Space-borne LWRA telescopes operating outside the ionosphere blanket avoid many of these problems associated with earth-based telescopes. Below frequencies of 4 or 5 MHz, space-borne LWRA telescopes are the sole means of obtaining LWRA measurement data.

To be useful, then, satellite-borne LWRA telescopes must, among other things, be able to operate at frequencies below 5 MHz, must be able to resolve to small angles for mapping, must be capable of monitoring time-varying sources, and be able to measure the polarization of the incident radiation. During the analysis and evaluation phases of the program, a listing of typical LWRA user requirements was developed as an aid in evaluating various satellite-borne telescope concepts; these user requirements are summarized in Table 4-10.

Table 4-10. Design Goals for a Long-Wave Radio Astronomy Antenna

Life Time:	Minimum of 1 year desired.
Orbit Altitude:	Minimum of synchronous.
Effective Beamwidth:	100 deg^2 at 1 MHz desired – less than 10° in one direction, but could be greater for solar and planetary astronomy. Interferometers should be used if possible for improving this resolution to 2°.
Pointing Accuracy:	1/2 beamwidth minimum to 1/10 for aspect determinations. In the case of a sweeping mode or drift mode antenna, the pointing direction must be known to within 1/10 half power beamwidth or better.
Pointing Stability:	Approximately 1/10 beamwidth or better.
Bandwidth:	500 kHz to 10 MHz desired, with emphasis on lower half.
Spectral Resolution:	Good desired, and depends only on electronics for any one antenna.
Sensitivity:	Unfilled apertures entirely adequate.
Lock-on-Time:	1/2 second to several hours for time varying phenomena. For most observations, however, an antenna arrangement with as slow a drift-rate as possible — of up to approximately 1 deg/sec suffices.
Tolerance:	Prefer 1/20λ, but 1/16λ is adequate. (At 1 MHz, $\lambda = 300$ m)
Orientation:	Eliminate antenna pattern directional ambiguity.

As reported in Volume II, various satellite structure concepts were evaluated for LWRA use. The evaluation showed that the variable geometry crossed-H interferometer concept best satisfied the LWRA user requirements, when considering the limitations imposed by structural and dynamic factors inherent in each of the concepts.

A most important feature of the crossed-H interferometer concept is its end-fire radiation pattern, which eliminates the hemispherical ambiguities in antenna response.

A second feature that makes the interferometer particularly useful is its variable tether length. This makes possible the use of the interferometer, together with data correlation processes, to achieve an unambiguous mapping resolution equivalent to that of a two-dimensional filled aperture array. This achieves a performance that could be matched, using conventional techniques, only by a vastly more complex antenna structure. A third very advantageous feature is the variable dipole spacing and length, which permits operation over the broad frequency range from 0.5 MHz to 10 MHz. A fourth important feature is the ability to lock-on or slew the end-fire dipole assemblies to continuously monitor one sector of the sky. This permits the use of the instrument to study time-varying sources, such as the sun, when events of special interest occur. The fifth advantage is that for strong time-varying sources, the entire range of 0.5 to 5.0 or 2.5 to 10.0 MHz can be used simultaneously by having one end tuned to 0.5 to 2.5 MHz or 5.0 to 10.0 MHz and the other end tuned to 2.5 to 5.0 MHz. A sixth advantage of the crossed-H configuration is that polarization measurements in two orthogonal directions are performed continuously during all observation periods and modes.

There are also several design features that played strong roles in the selection of the crossed-H interferometer, such as deployment reliability, refurbishment capability, cost considerations, etc. These are discussed in other sections.

4.5.2 <u>Operational Characteristics of the Crossed-H Interferometer.</u> Operational characteristics of the variable-geometry crossed-H interferometer and a comparison with user requirements are summarized in Table 4-11. The crossed-H interferometer LWRA capabilities include two principal missions or modes of operation: a high resolution mapping survey of background and discrete source radiations over the entire celestial sphere and monitoring of the radiation from strong time-varying sources. The mapping mission is the most demanding in terms of structural, dynamic and instrumentation considerations, as well as the time required to complete it. Typical mission profiles for the mapping (source survey) and strong time-varying source observation modes are shown by Figures 4-30 and 4-31.

Figure 4-30 shows the primary survey procedure that will map the sky at frequencies between 0.5 MHz and 5.0 MHz, requiring 333 days of mapping time. It also shows an optional procedure that extends the mapping to frequencies up to 10 MHz, which requires a total of 428 days. This option is an example of varying the missions that might be scheduled. The value of using satellites for mapping above 5.0 MHz is questioned

Table 4-11. Operational Characteristics of the Crossed-H Interferometer and Comparison with User Requirements

FEATURE	SPECIFIED USER REQUIREMENT	CROSSED-H INTERFEROMETER CHARACTERISTICS
Lifetime	Minimum of one year	Meets requirement
Orbit Attitude	Minimum of synchronous	Meets requirement
Effective beamwidth	100 $\deg^2$ at 1 MHz desired, interferometer if possible to improve resolution to $2°$	Beamwidth $225° \times 90°$ at 0.5 MHz with interferometer resolution in one plane to $1.7°$
		Beamwidth $180° \times 40°$ at 2.5 MHz and above with interferometer resolution in one plane to less than $35°$
Pointing Accuracy	1/2 beamwidth minimum to 1/10 beamwidth	Meets requirement
Bandwidth	0.5 to 10 MHz	Meets requirement
Spectral Resolution	Good desired, depends on electronics	Meets requirement
Dynamic Range	Unfilled apertures are adequate	Meets requirement
Lock-on Time	1/2 second to several hours for time varying phenomena	Meets requirement
Tolerance	Prefer $1/16\lambda$, $1/10\lambda$ is adequate	Meets requirement

by many in the radio-astronomy community, because of the high probability of interference from man-made and atmospheric radiation and also because some earth-based measurements in the 5.0 MHz to 10 MHz frequency range are possible. It is considered, however, that the capability for making such mapping measurements should at this stage be included as part of the crossed-H interferometer characteristics. This capability can be included at a relatively small cost and will permit measurements in the 5.0 to 10 MHz frequency range at such times (or under such conditions that) when earth-based radio telescopes are useless because of ionosphere reflections and refraction.
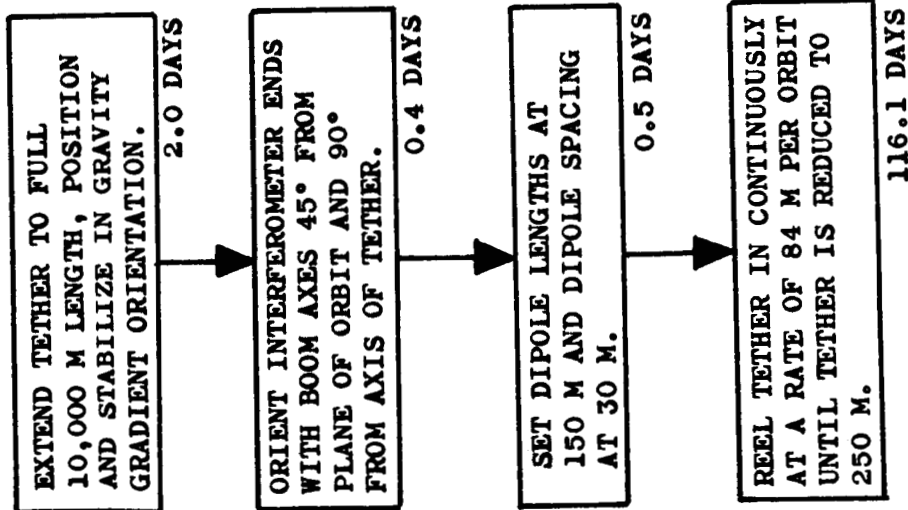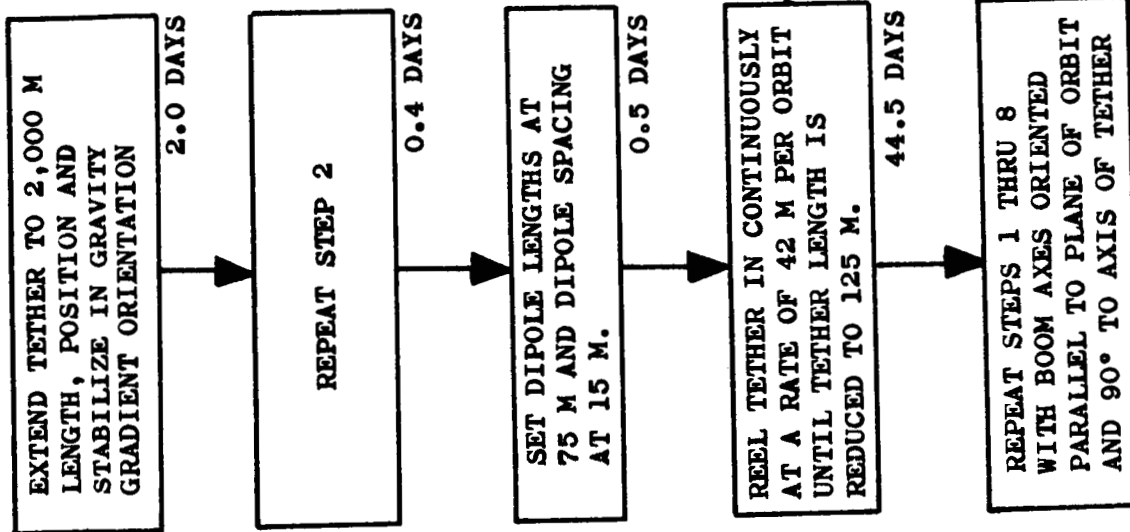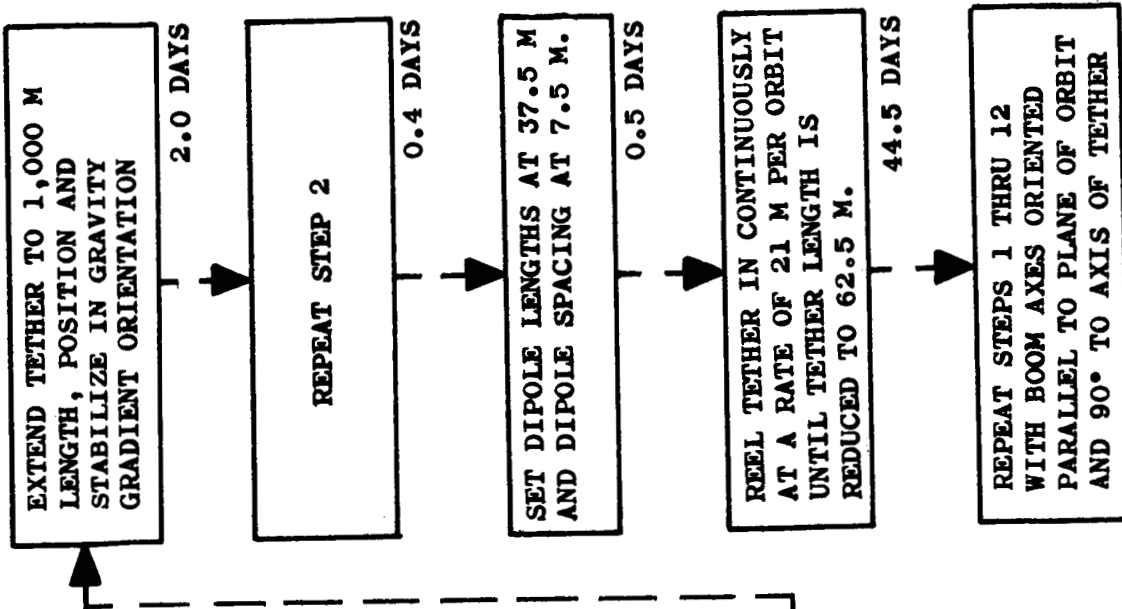
Figure 4-30. Assumed Source Survey Mission

```
┌─────────────────────────────────────────┐
│ WITH TETHER SET AT ANY CONVENIENT       │
│ LENGTH, ORIENT INTERFEROMETER ENDS      │
│ WITH BOOM AXES IN DIRECTION OF          │
│ TARGET SOURCE.                          │
└─────────────────────────────────────────┘
                                  240 MIN.

┌─────────────────────────────────────────┐
│ SET DIPOLE LENGTHS AND DIPOLE SPACING   │
│ OF ONE SATELLITE FOR THE 0.5 MHz TO     │
│ 2.5 MHz BAND AND THE DIPOLE LENGTHS     │
│ AND SPACING OF THE OTHER SATELLITE      │
│ SET FOR THE 2.5 MHz TO 5.0 MHz RANGE    │
└─────────────────────────────────────────┘
                                   20 MIN.

┌─────────────────────────────────────────┐
│ CONTINUOUSLY ORIENT INTERFEROMETER      │
│ ENDS SO THAT BOOM AXES CONSTANTLY TRACK │
│ IN DIRECTION OF TARGET SOURCE FOR       │
│ PERIOD OF MEASUREMENT                   │
└─────────────────────────────────────────┘
                                    X MIN.

┌─────────────────────────────────────────┐
│   REORIENT AND REPEAT FOR NEXT TARGET   │
└─────────────────────────────────────────┘
```

TOTAL TIME PER SOURCE X + 260 MIN.

Figure 4-31. Strong Time Varying Source Observation Mode

As Figure 4-30 shows, the source survey procedure uses three different maximum tether extensions that decrease in length as the frequency band increases. Thus, the maximum tether length for the 0.5 to 2.5 MHz band is 10,000 meters; for the 2.5 to 5.0 MHz band it is 2000 meters; and for the 5.0 to 10 MHz band it is 1000 meters. These maximum lengths are those that will make the resolution perpendicular to the plane of the orbit equal to or less than $1.7°$ for all of the frequencies in each band. The best angular resolution as a function of frequency for this set of tether lengths is shown by the solid lines of Figure 4-32. Also shown are the mapping times required in each band. The dashed lines of Figure 4-32 show other best angular resolutions possible, with their corresponding total mapping time requirements, using maximum respective tether lengths for the next higher bandwidth. It may be seen how increased resolution requires correspondingly longer survey times. While the resolution shown by the dashed lines can theoretically be obtained, the mapping times may become excessive; for that reason, the tether lengths shown in the procedures of Figure 4-30 are considered to be an optimum compromise between high angular resolution and long mapping times.

The angular resolutions achieved by the interferometer in the mapping mode are obtained by using correlation techniques described by Brown and Lovell (Reference 7) and others to synthesize the directional performance of a filled aperture array. A summary of this aperture synthesis is given in Table 4-12. Figure 4-33 illustrates the resolution obtainable at various angles for a single mapping; that is, without repeating the mapping after the orbit plane has precessed. This is identical to the resolution possible with a filled aperture antenna having a diameter equal to the maximum tether length of the interferometer.

The variable geometry features of the crossed-H interferometer are indispensable for the effective employment of the interferometer as a space-borne radio telescope. The adjustments normally made directly on earth-based telescopes must, in the case of the orbiting telescope, be remotely controlled from earth. The crossed-H interferometer performs, by remote control, the operations required for a particular measurement. It also has, as a result of its variable geometry, the flexibility necessary for making measurements over the large desired frequency range. The tether length must be varied to perform the aperture synthesis mapping mission. Also a requirement for the strong time-varying source mission is the ability to slew the end-fire pointing of the dipole arrays so as to eliminate ambiguities caused by reception from unwanted directions. The variable dipole lengths and spacings permit sustaining a coherent antenna pattern throughout the frequency range of 0.5 to 10 MHz.

4.6 THERMODYNAMICS ANALYSIS

4.6.1 Environment

4.6.1.1 Launch. During launch the spacecraft is shrouded by the launch vehicle fairing and thus protected from direct exposure to the environment. However, the fairing walls are directly heated and transmit the energy to the spacecraft by radiation.
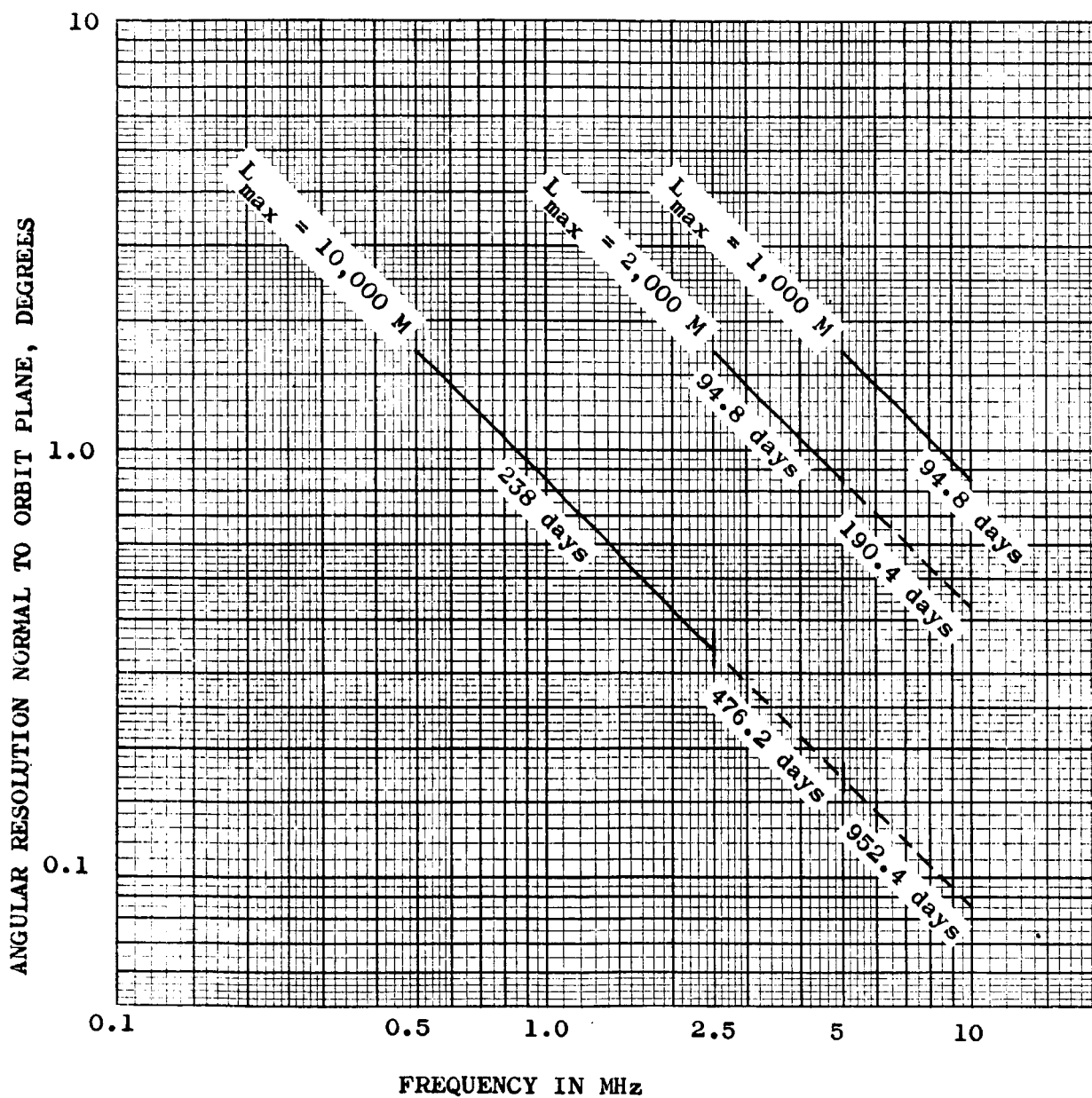
Figure 4-32. Mapping Resolution as a Function of Frequency for Various Maximum Tether Lengths and Mapping Times
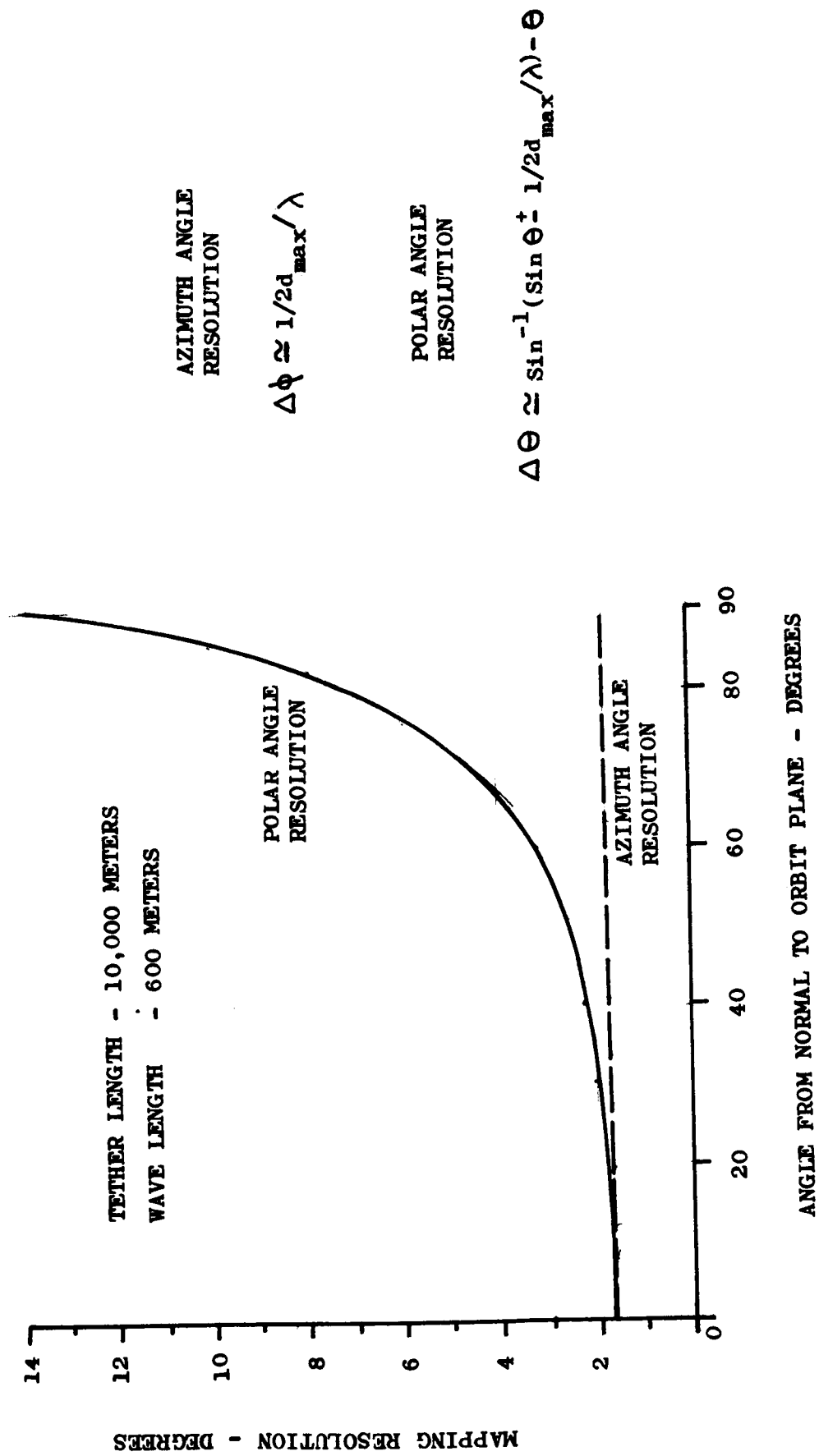
Figure 4-33. Mapping Resolution vs. Angle from Orbit Plane

Table 4-12. Aperture Synthesis Summary

For synthesis it is assumed that all sources have constant mission during the mapping period.

A complete map of the sky at 15 frequencies between 0.5 and 10 MHz requires 428 days.

Angular resolution obtained is:

$1.7° \times 1.7°$ (2.9 deg^2) or better normal to orbit plane

$1.7° \times 2.6°$ (4.5 deg^2) or better at 45° to orbit plane

$1.7° \times 10°$ (17 deg^2) or better in orbit plane.

Improved resolution, especially at the higher frequencies, may be achieved by using larger maximum tether lengths during the aperture synthesis. Improved resolution may also be achieved by repeating the mapping process after the orbit has precessed. By three such mappings, resolution can be improved to $1.7° \times 2.6°$ (4.5 deg^2) or better over the entire sky.

This thermal load can be minimized by proper surface conditioning of the inner surface of the fairing, such as with an aluminum coating.

A detailed thermodynamic analysis would establish the requirements of air conditioning prior to liftoff and the necessity of a thermal shroud during ascent.

4.6.1.2 Orbit. The synchronous orbit in which the spacecraft operates results in a very small view factor of the earth disk, about 0.023; correspondingly low values of albedo, 4 Btu/ft^{2}hr; and earth thermal, 1.7 Btu/ft^{2}hr. A good approximation to the actual incident energy would be just the direct solar heating of 442 Btu/ft^{2}hr. Although the albedo and earth thermal radiation add to the heat load, they generally tend to offset distortion and, therefore, use of only direct solar heating is slightly conservative.

Some of the thermal calculations require evaluation of the residence time of the spacecraft within the earth's shadow. For the 28.5° inclination, the calculated maximum shade time is 31.3 minutes each 24 hours.

4.6.2 Thermodynamic Distortion. Exposure to the orbital environment results in non-uniform heating of structural members. This derives from the variation of illumination (absorbed unit energy) with surface aspect angle and the independence of surface radiation from angular effects. The resulting variation in the net heat balance with local surface position establishes temperature gradients throughout the material.

Thermodynamic distortion results from differential expansion of the material in response to these temperature gradients. The extent of this distortion was evaluated for the telescoping boom and for the dipoles. Distortion of these members can cause misalignment of the antenna elements.

4.6.2.1 Boom. Maximum thermal gradients were calculated for the extending boom to evaluate distortion limits and to assist in the design of material thickness, web location, and surface treatment. These calculations were based on an assumed orientation with respect to the ecliptic, which would result in the most critical illumination of the boom surfaces. This orientation is shown in Figure 4-34.

Two separate cases were considered. The first case assumed the web pattern on all three faces of the boom were the same. Thus, the orientation shown would result in two faces being shadowed. In the second case it was assumed that the web patterns were staggered, and thus the inside of second face would also be illuminated.

Steady state calculations were made on the computer by modeling a thermally symmetric portion of the boom. This was done by dividing the media into 11 nodes. Heat balance at each node was computed from conduction to adjacent nodes, illumination from the sun, and radiation to the space environment. Two surface treatments were considered: anodized with $\alpha_s = 0.92$ and $\epsilon_T = 0.84$; and white paint with $\alpha_s = 0.16$ and $\epsilon_T = 0.85$. Resulting steady state temperature of the aluminum structure is given in Table 4-13.

Deflection of the tip of the boom was calculated from

$$\delta = \frac{\epsilon \ell^2 \Delta T}{2D}$$

where

δ = tip deflection

ϵ = linear coefficient of expansion

ℓ = length of boom

ΔT = controlling temperature gradient

D = controlling length

Asymmetry of the calculated temperatures with respect to the boom geometry results in deflection about a skewed plane. The direction of deflection differs for the aligned and staggered cases and is shown in Figure 4-35. Variation of tip deflection with boom extension is shown in Figure 4-36.

The angular displacement of the end of the boom, and thus the misalignment of the dipole planes, is equivalent to the tangent angle of the deflected tip.

4-88

9
2
11
8
1, Node
10
6 4
7
5 3

Sunlite

12"

0.09x1.0, Web

0.032x0.5, Web

1.0x2.0, Cap

11"
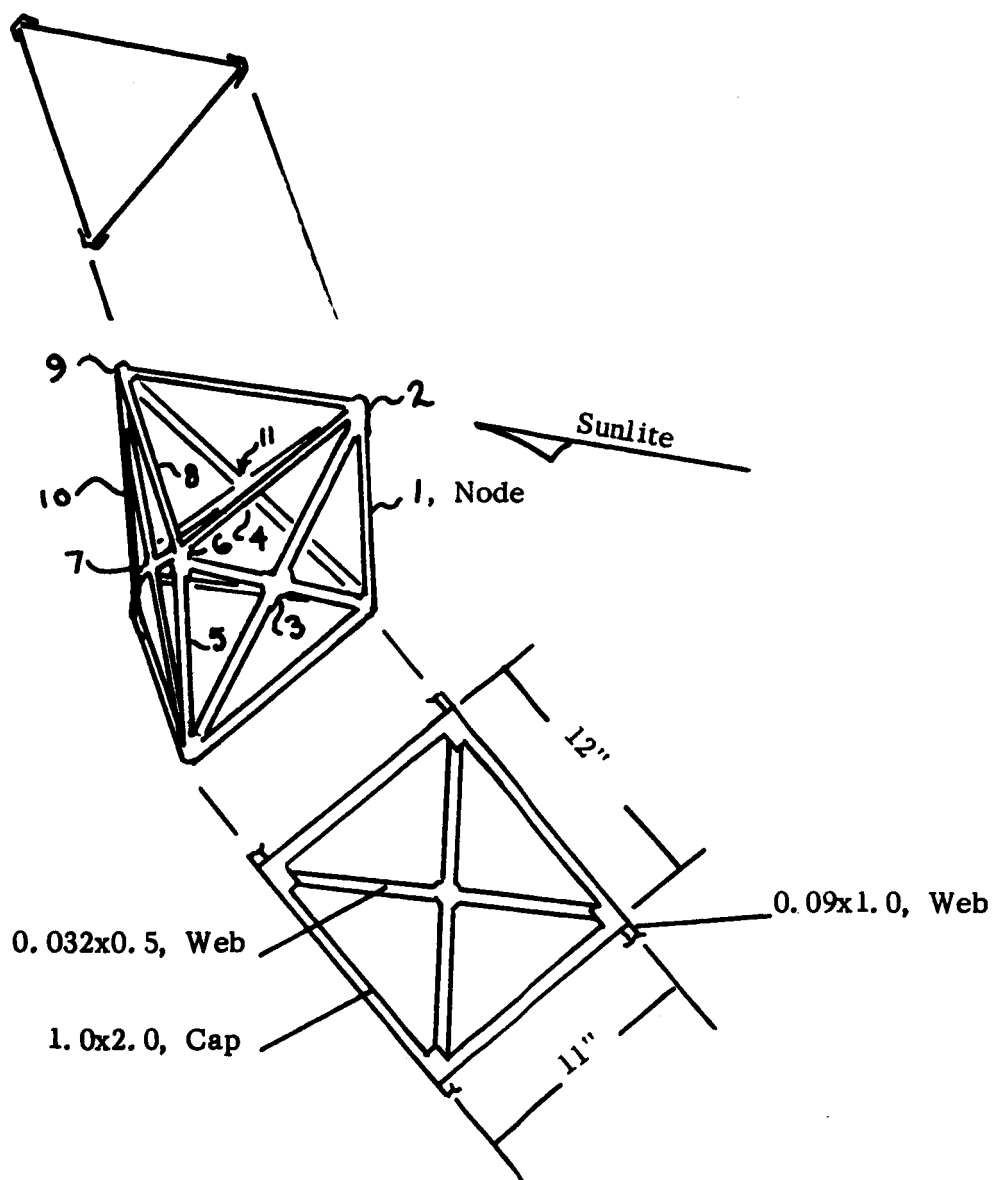
Figure 4-34. Extending Boom Orientation

Table 4-13. Extending Boom Steady State Temperatures
(See Figure 4-34 for Nodal Position)

| NODE | ANODIZED (°R) | | WHITE PAINT (°R) | |
	ALIGNED	STAGGERED	ALIGNED	STAGGERED
1	444	495	276	287
2	447	454	276	289
3	559	562	322	345
4	500	511	287	308
5	466	483	276	306
6	476	499	276	309
7	340	553	234	328
8	-	487	-	297
9	317	385	226	263
10	308	368	223	258
11	330	352	234	253

$$\theta = 2 \tan^{-1} \frac{\delta}{\ell}$$

The variation of this value with boom length is shown in Figure 4-37. Since each half
of the boom is mounted asymmetrically (Figure 4-35), the direction of deflection afford
some compensation to the dipole plane misalignment. The benefit of this compensation
is given in Table 4-14.

Table 4-14. Reduced Angular Displacement (White Paint, ℓ = 15 m)

	NO COMPENSATION	COMPENSATION
Aligned Members	2.48	2.15
Staggered Members	1.86	1.10
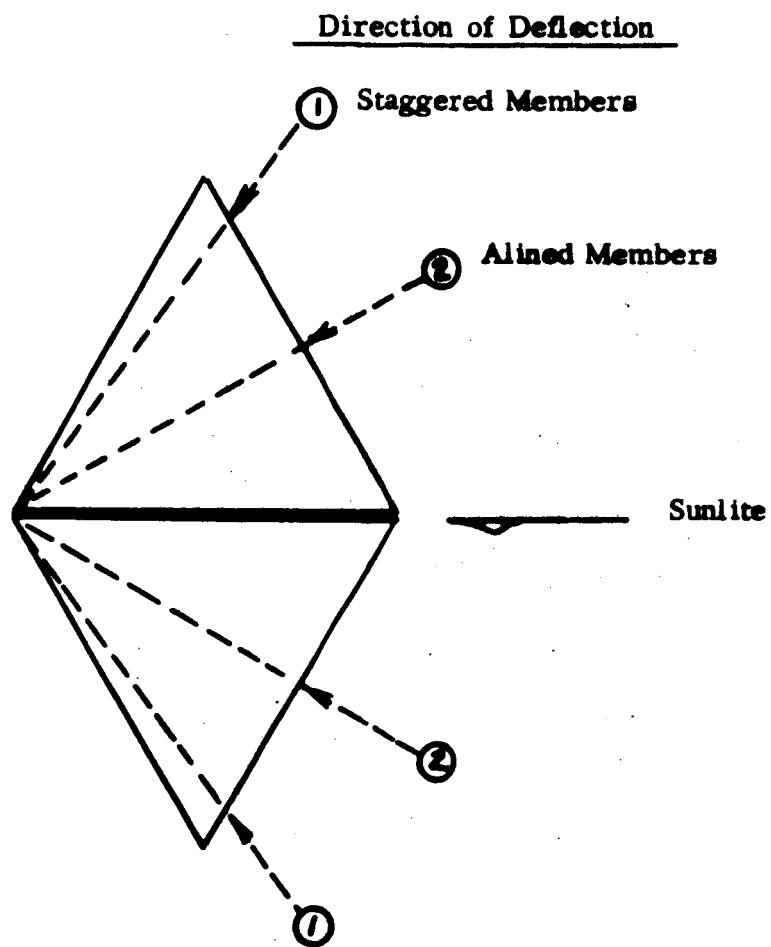
Figure 4-35. Direction of Boom Deflection
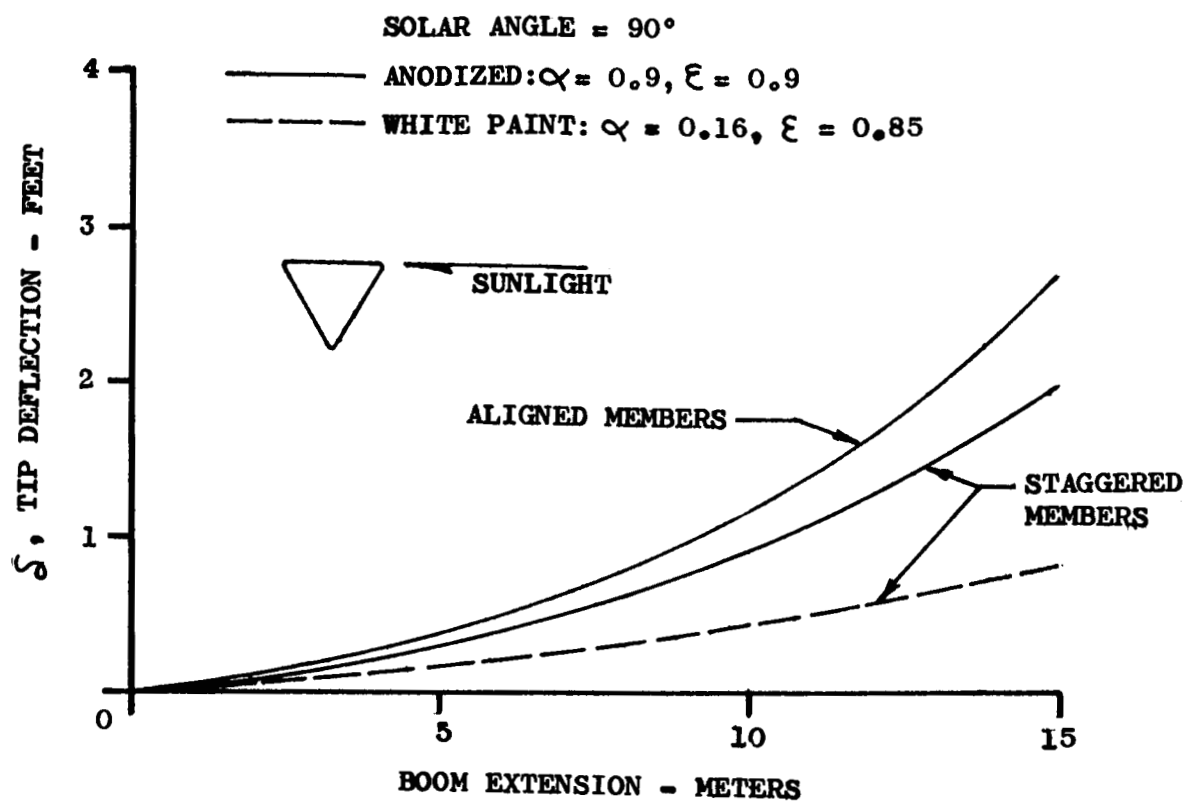
Figure 4-36. Variation of Tip Deflection with Boom Extension
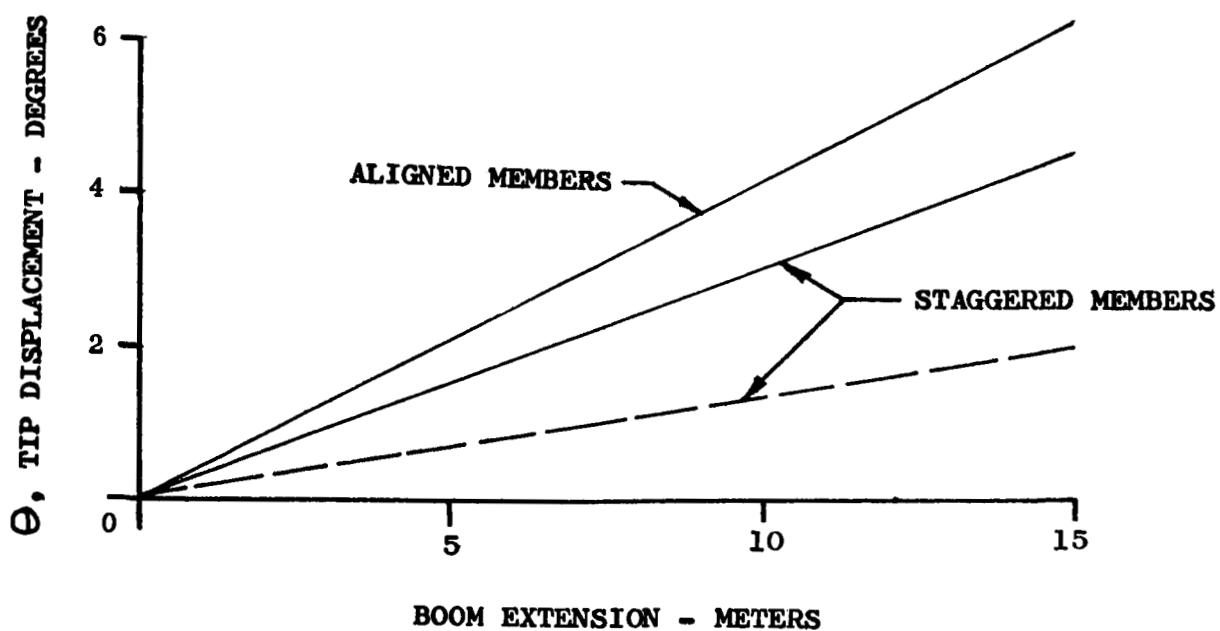
Figure 4-37. Variation of Angular Displacement with Boom Extension

4.6.2.2 _Dipoles_. Previous analysis at Convair (Reference 8) revealed the advantage of screen type tubular elements in contrast to solid surface type tubes. Use of the screen material allows illumination on the interior of the trailing cylindrical surface, thereby minimizing temperature gradients. In addition, anisotropic materials can be used for the longitudinal wire and the circumferential wires to minimize thermal expansion and maximize thermal conduction, respectively.

Initial calculations were made to evaluate the effect of tube diameter on distortion. The wire screen tube parameters used for this study are given in Table 4-15. Invar with its low coefficient of thermal expansion was used for the longitudinal wires, and beryllium-copper with its high thermal conductivity was used as the circumferential wires. Results of these calculations are shown in Figures 4-38 and 4-39 for the tip deflection and displacement, respectively. Here, the angular displacement is defined as $\theta = \tan^{-1} \frac{\delta}{L}$. Two surface conditions were investigated; bare metal with $\alpha = 0.4$ and silver plated with $\alpha = 0.05$. A substantial reduction in distortion is obtained with silver plating. Previous studies revealed the dependence of distortion on solar angle, ϕ, on the seam location. These studies revealed that $\phi = 10°$ would result in the worst "self-shadowing" of screen circumferential wires. A greater effect on distortion results from the influence of the seam location on the temperature gradient. In this case it is assumed that no heat transfer occurs across the seam, and that the seam has no overlap.

Variation of tip deflection with mesh size and wire diameter was calculated and is shown in Figures 4-40 and 4-41, respectively, for a 2.0-in.-dia. tube. The mesh size contributes to a wide variation of deflection due to the reduction of illumination shine-through to the trailing surface as the mesh density increases.

Table 4-15. Wire Screen Tube Parameters

DIAMETER (in.)	MESH (in.^{2})	LONGITUDINAL WIRE DIA. (in.)	CIRCUMFERENTIAL WIRE DIA. (in.)
0.5	12 X 12	0.009	0.009
1.0	12 X 12	0.009	0.009
2.0	7 X 7	0.015	0.015
4.0	5 X 5	0.020	0.020

Note: Longitudinal Material: Invar
Circumferential Material: Beryllium-Copper
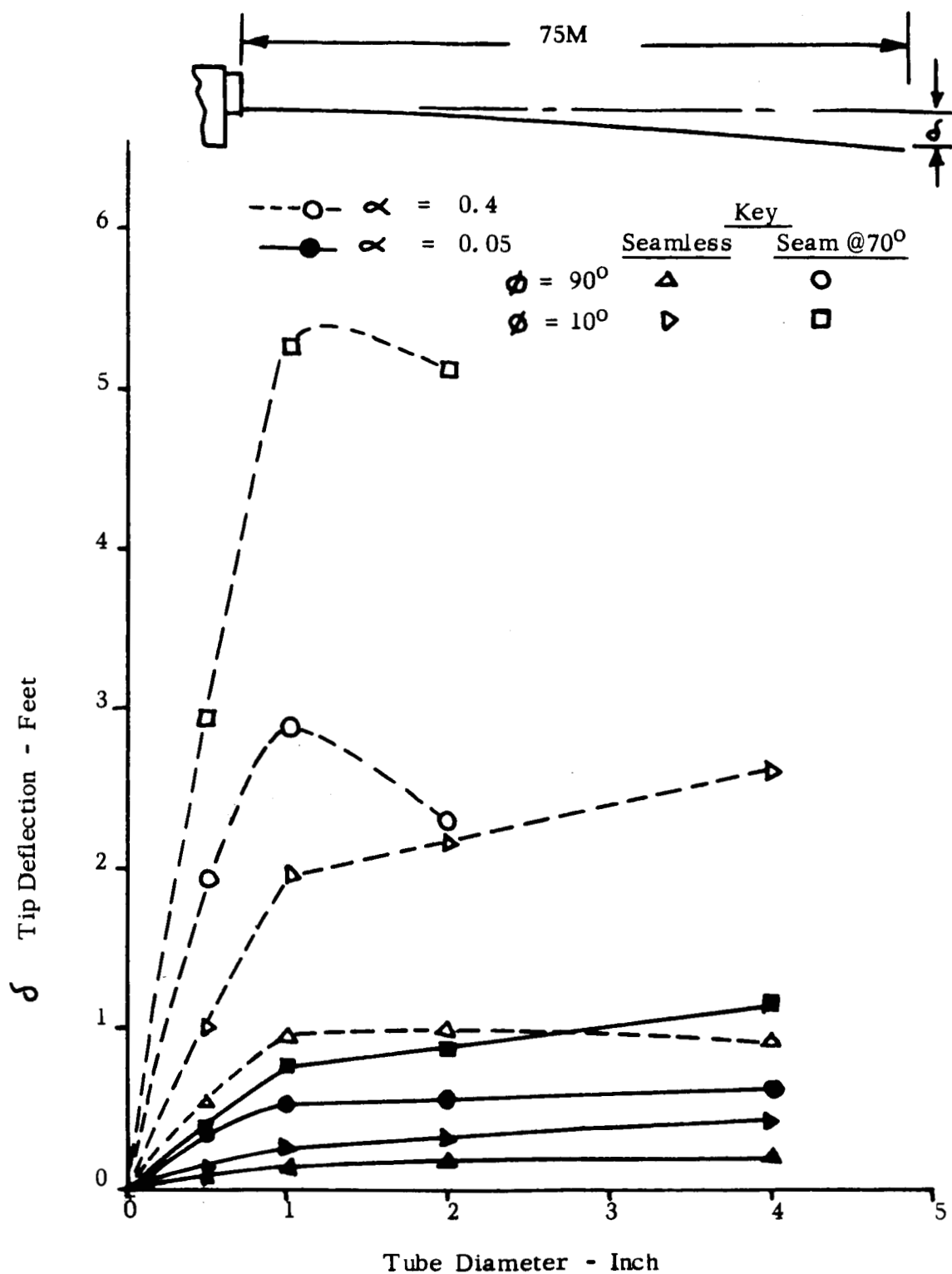
Figure 4-38. Deflection of Half Dipole at Full Extension

4-94
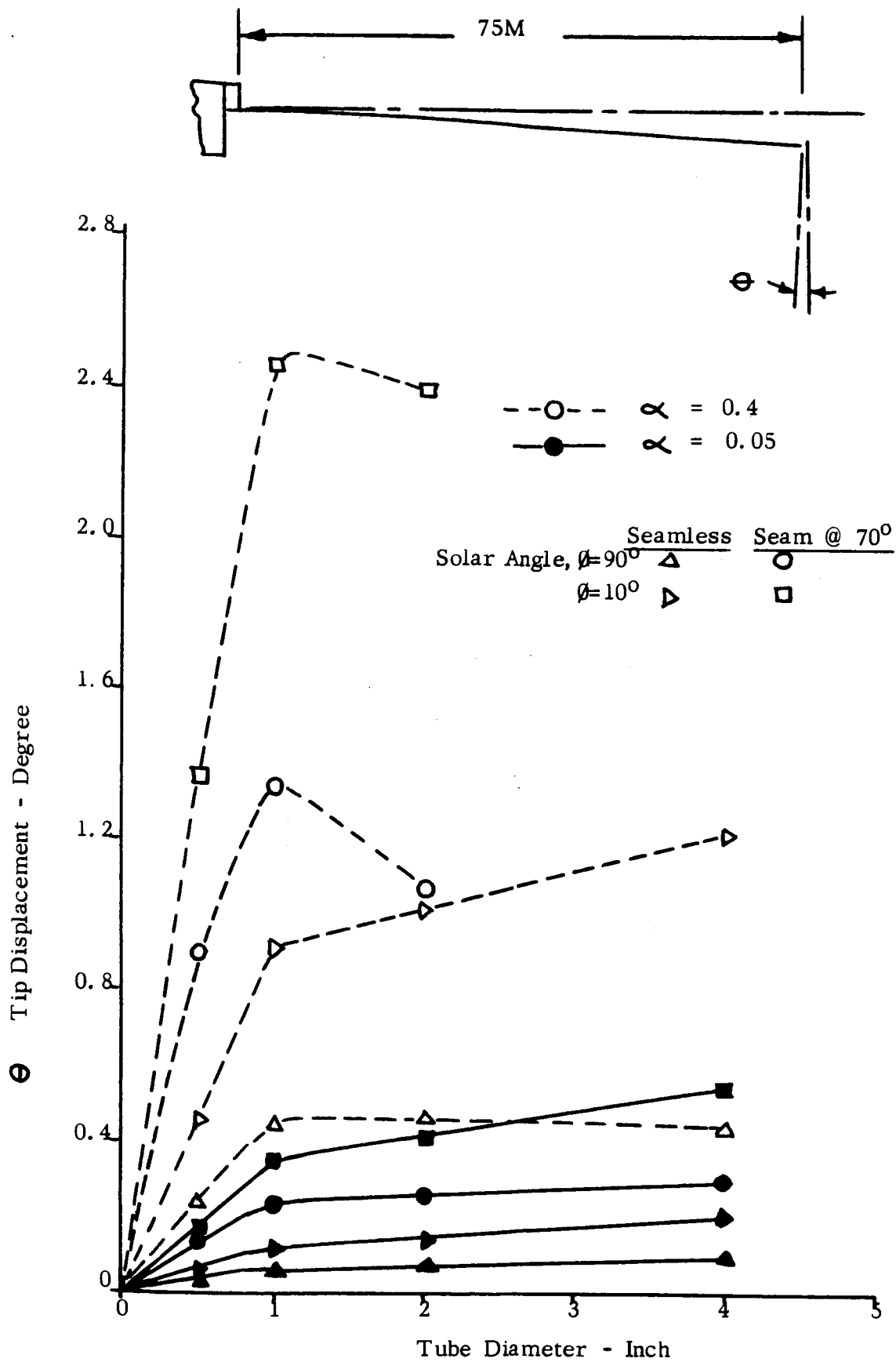
Figure 4-39. Displacement of Half Dipole at Full Extension
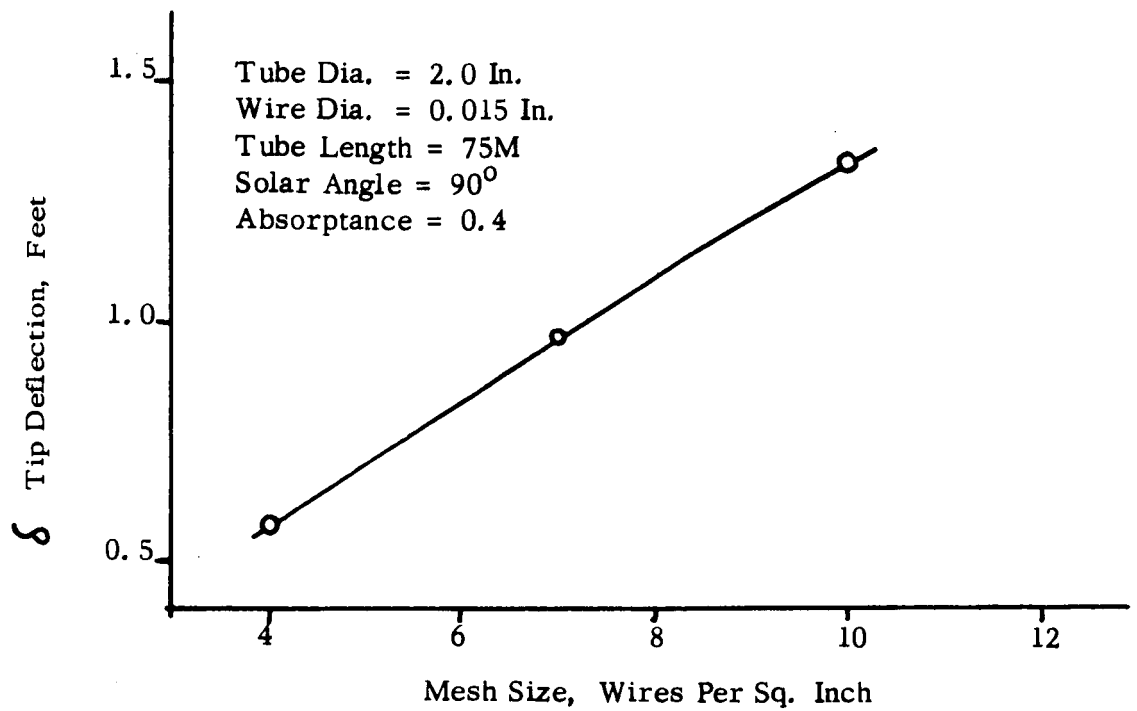
Figure 4-40. Variation of Seamless Tube Deflection with Mesh Size
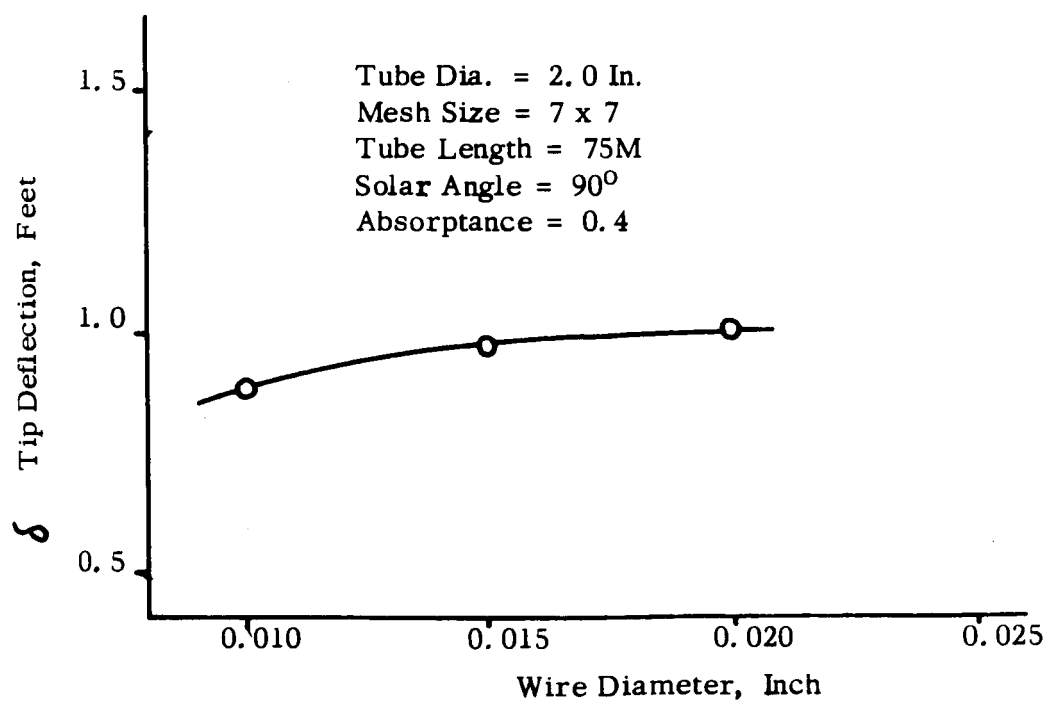
Figure 4-41. Variation of Seamless Tube Deflection with Wire Diameter

The effect of a 180° seam overlap was calculated for the 2.0-in.-dia. tube at a 90° solar angle. In these calculations it was assumed that the overlap would place twice as many wires in the path of the illumination and thus cause the greatest possible shadowing of shine-through to the trailing surface. Figure 4-42 gives the results and indicates a variation of about three in the temperature gradient between the seam being at the leading surface or trailing surface.

Results of the generalized wire screen distortion studies were scaled to the dipole design of the crossed-H interferometer. This design consists of 0.009-in. wires in a 12 X 12 mesh using Elgiloy for the longitudinals and beryllium-copper for the circumferentials. Silver plating was assumed for the surface treatment. Calculated tip deflection and displacement are shown in Figures 4-43 and 4-44, respectively. The worst possible case would result if the seam were at 70° and a solar angle of 10°. While an orientation resulting in this solar angle is a realistic possibility, the seam position varies in a spiral fashion along the length. Therefore, the true displacement would probably be represented by an average between the seamless and the 70° seam position results.

4.6.3 Subsystem Environmental Control Requirements. The spacecraft subsystems are contained within the main body volume. Steady state temperatures were estimated for operation in the sun. Actual operating temperatures depend on a detailed radiation/conduction analysis of the heat flow between internal components and the main-body housing. This type of analysis depends on specific information as to component arrangement, heat output, surface conditioning, mounting, etc., and could not be realistically modeled at this time.

An approximate analysis was made by assuming that good heat interchange could be achieved between internal equipment and the main-body housing. Furthermore, it was assumed that the illuminated side of the housing could radiate to the opposing side without appreciable shadowing. Using a thermal emittance of about 0.9 for surfaces within the main body, nearly uniform temperatures result.

The external surface of the main body is nearly covered with solar cells for which $\alpha_s = 0.925$ and $\epsilon_T = 0.843$ were taken. Calculations revealed that the absorbed solar energy is about one hundred times that dissipated by the internal equipment. This results in a uniform temperature of 66°F when the solar vector is at a right angle to the boom or 40°F when aligned with the boom.

Cool down during transit through the earth's shadow was calculated assuming the equipment weighed 1000 pounds and had an average specific heat of 0.2 Btu/lb°R. Using a thermal emittance of 0.84 the average temperature drop would be less than 10°F, and thus the equipment would emerge from the shade at about 30°F.
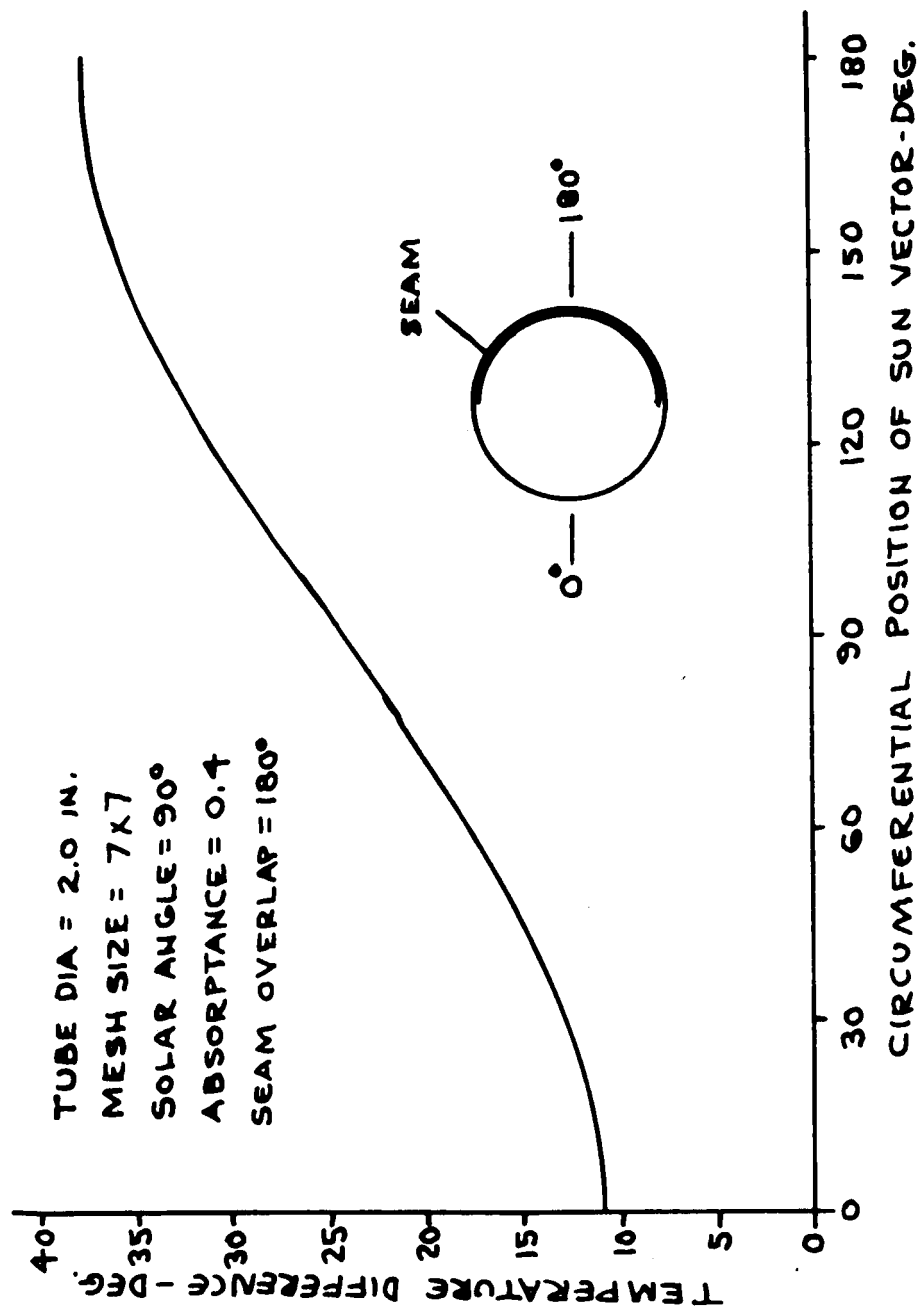
Figure 4-42.   Effect of Sun Angle on Temperature Difference, 180° Seam
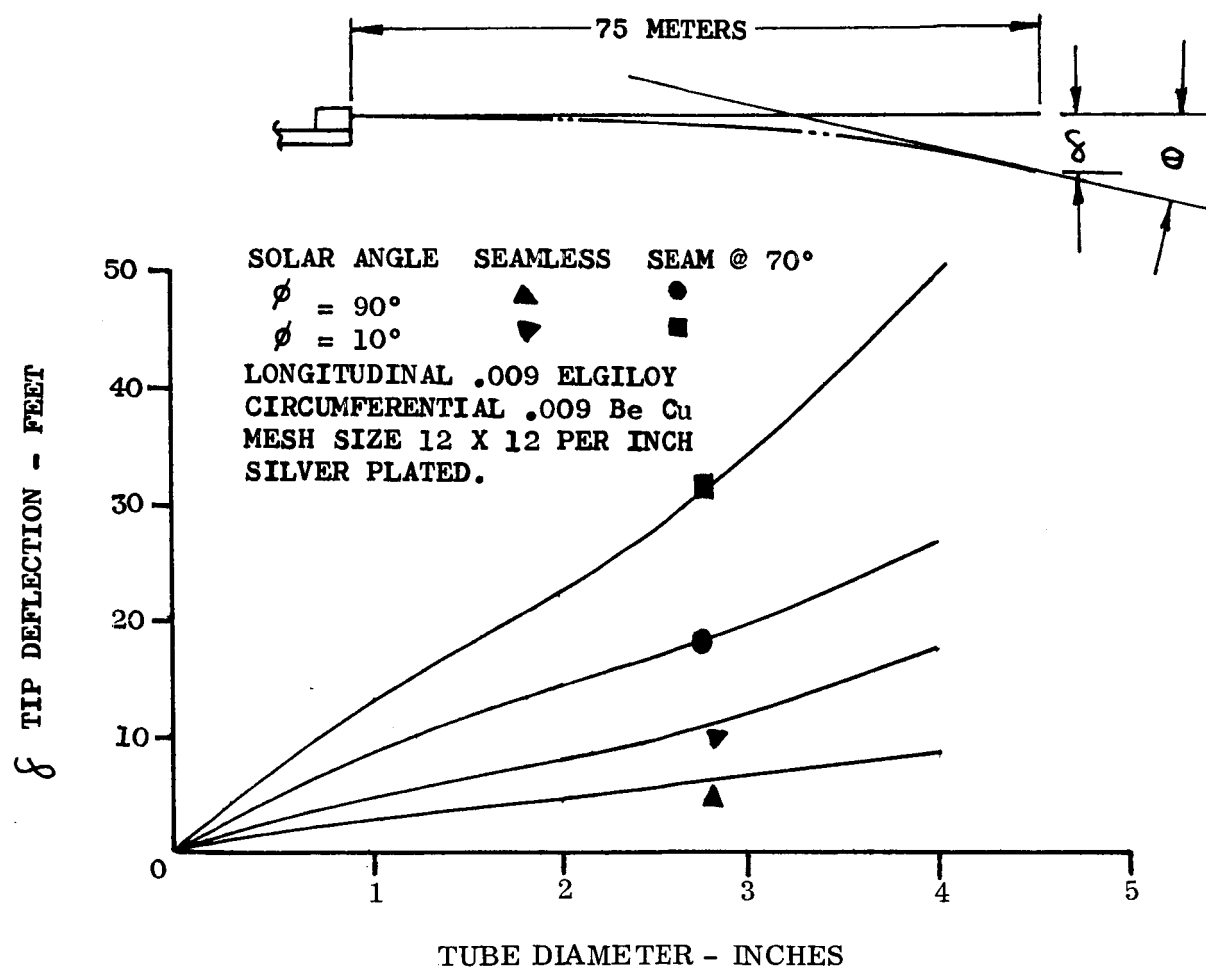
Figure 4-43. Deflection of Half Dipole at Full Extension
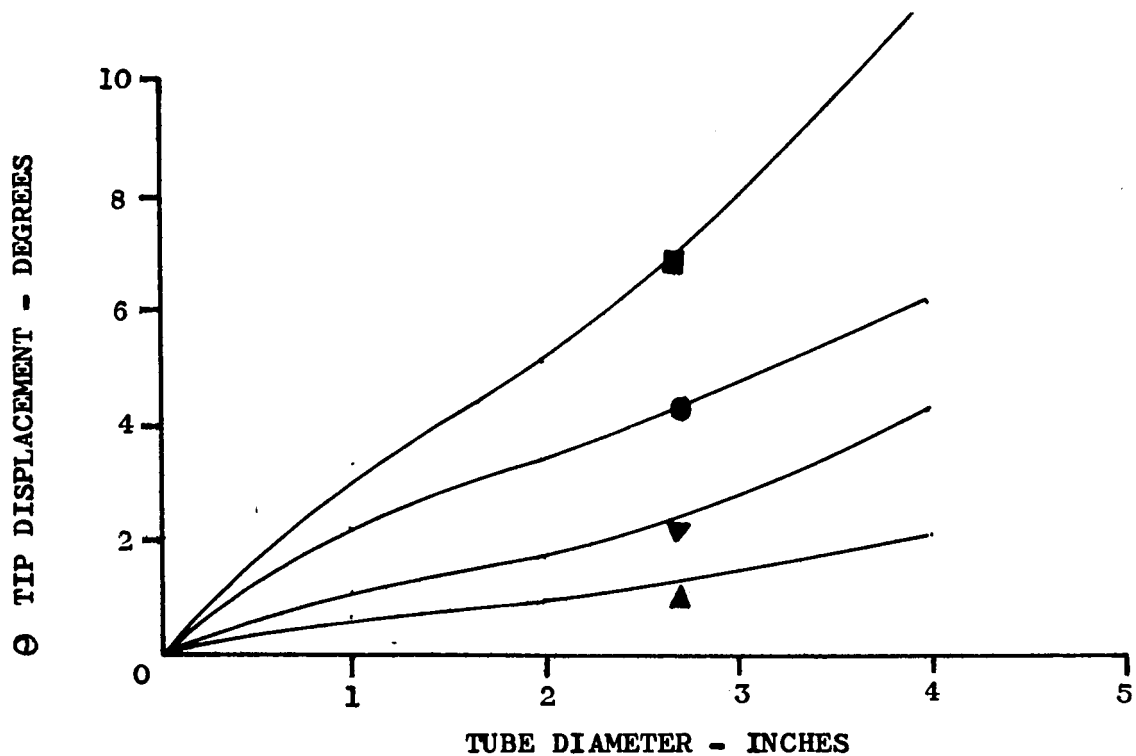
Figure 4-44. Displacement of Half Dipole at Full Extension

SECTION 5

CREW SYSTEM CAPABILITIES AND INFLUENCE ON DESIGN

The capabilities of man in space are a significant factor in the achievement of mission success. The astronaut is used to advantage both within the CM and in EVA. Many of the functions he performs could be done automatically or by telemetered command, but at some penalty in weight, cost, or reliability. The additional functions he performs could not be performed in other ways that are feasible in the state of the art for the immediate future.

The capabilities of the astronaut have certain limitations that must be evaluated for each particular task, and the design of the operation and the article modified as necessary.

One of the most significant of these limitations is the relatively short period available each day for task accomplishment in EVA. This is reflected in the checkout operations where the tests are accomplished by command from within the command module (or ground station) and the results recorded in the same manner plus visual supervision. Only in the event of malfunction is the astronaut required to leave the shirt-sleeve environment of the CM.

Crew systems constraints shall be based on the NASA/MSC document entitled "Summary Description of Baseline Extravehicular Astronaut for 1968-1972," report number CSD-S-012, dated February 21, 1967, also included in Volume II, Appendix B, of this report. This document provides a description of an extravehicular astronaut and his equipment for the time period of 1968 to 1972 or mainline Apollo and early Apollo Applications Program extravehicular activity missions. The baseline astronaut's capability is based on utilization of equipment that is presently available or now under development for use during the 1968-1972 time period.

In the event of such a malfunction, the advantage of having man available for immediate repair or replacement is obvious. To eliminate the dangers of the astronaut's working on an adjacent structure, not integral with the CM, a docking ring is provided on each satellite. With the CM docked to the satellite and the booms retracted by command, the distance the astronaut has to traverse is minimized, thereby saving time and energy and increasing the safety factor. Under such conditions, an AMU is not required.

Further provisions for the astronaut's convenience include hand rails, stirrups, and anchor points as indicated in Figure 5-1. These will facilitate the use of tools and EVA equipment such as fixed harness support and dutch shoes. Access is provided by 30 × 30-in. hinged panels on the center body that open up to reveal equipment mounted on the boom sections as well as the electronics mounted on the inside of the panels.

HAND & FOOT
RESTRAINT RAILS

DUTCH SHOE
RESTRAINT

CSM

VARIABLE WAIST WORK
SITE RESTRAINT

TETHER

PLSS UNIT
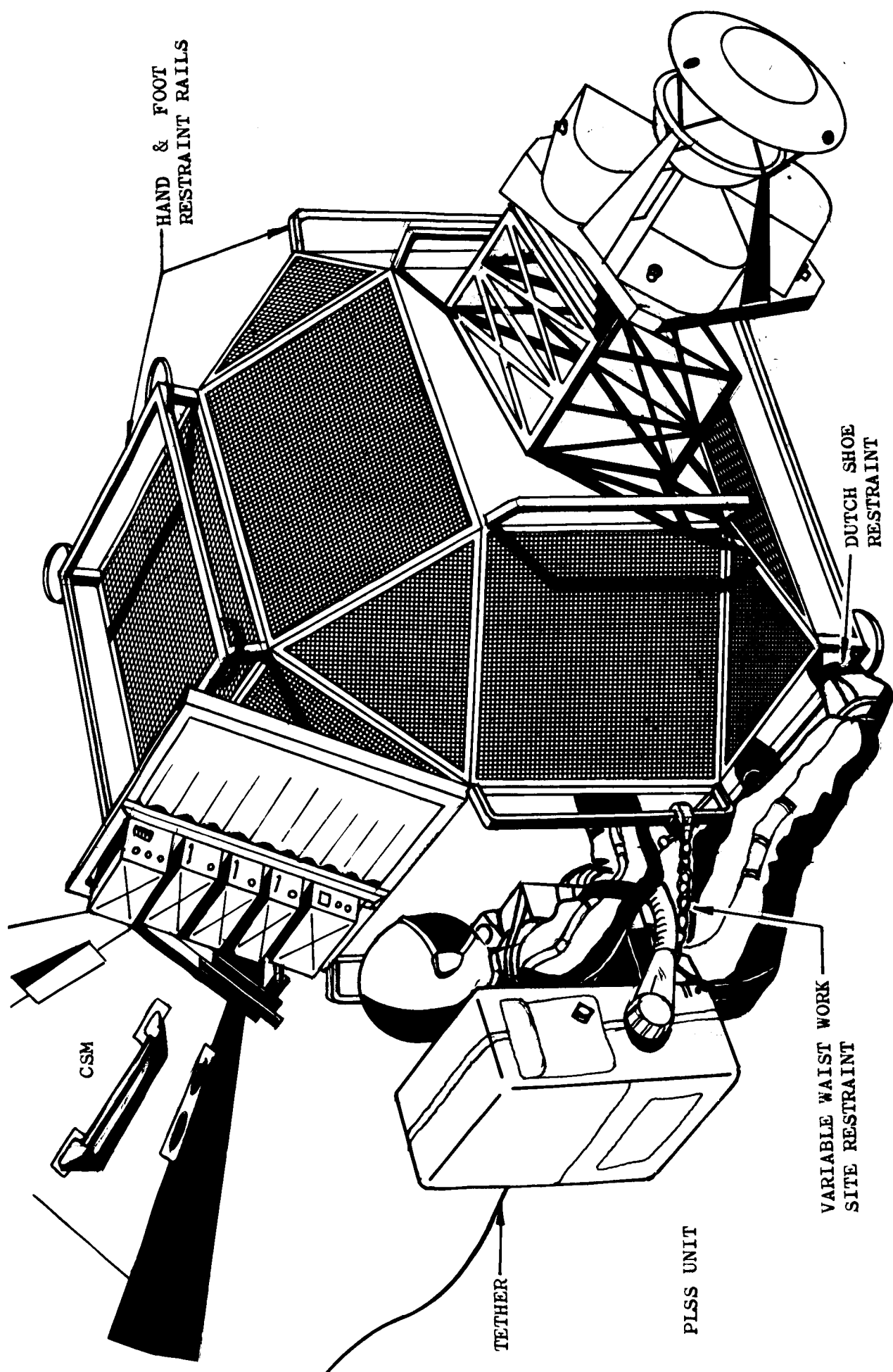
Figure 5-1.   Center-Body EVA Activity

Replaceable modules are mounted with gross fasteners; such as, toggle clamps or pull pins. Where possible, the old modules are left in place to save astronaut energy otherwise spent in disposing of them; e. g. , batteries and solar cell panels.

Equipment that protrudes, such as telemetry antennas, are made flexible and are rubber coated to prevent harm to the astronaut or his suit. The dipoles and booms are always retracted before docking and EVA to prevent harm to the structure as well as to the astronaut.

5.1 OBJECTIVES ACHIEVED BY EVA. The normal deployment and operation mode of the crossed-H interferometer is automatic. However, EVA is required for observation and inspection, maintenance, repair, refurbishment, and updating of components. It is through these functions that man will develop the EVA capability of supporting future space missions. As previously discussed, the first flight objective is to evaluate the role of man in these functions. Successful operation of the antenna will be enhanced by the following EVA:

a. The repair and/or replacement of any components that are malfunctioning.

b. Extending the life of the experiment by snapping in new solar cell panels and dipole modules over the expended ones. Also replenishing the propellant for the attitude control system, installing new ACS valving as required and installing new batteries.

c. Updating the antenna components in orbit with the advancing state-of-the-art.

d. Observing conditions on board the satellite that precede a failure before such failure occurs and performing remedial operations.

5.2 CREW TASKS AND TIME-LINE ANALYSIS. This functional flow diagram (Figure 5-2), an arbitrary sample, and resultant time-line analysis, (Table 5-1) reflect EV activity to extend the life of the crossed-H interferometer radio astronomy antenna by refurbishment. Table 5-1 represents a time distribution for an average annual refurbishment operation in which the following components are replaced;

a. Eight solar cell panels.

b. Six dipole modules.

c. Refill propellant tanks.

The above components reflect the total replacement parts for both satellites, which are docked together for the refurbishment operation.

The total operation for the task outlined above shows the requirement of 3 days to complete the refurbishment with Astronaut A working a total of 8.66 hours in EVA activity while Astronaut B works a total of 44 minutes in an EVA mode. The EVA

Figure 5-1 — Flow diagram boxes:

- DOCK THE CM TO THE MAIN BODY OF SATELLITE I
- PERFORM PRE-EVA. (ASTRONAUT A&B)
- OPEN HATCH & EXIT FROM CM (A&B)
- MANEUVER TO STOWAGE AREA AND ANCHOR TO A SUITABLE STRUCTURE (A&B)
- UNSTOW CLOTHESLINE RIG AND ATTACH TO STRUCTURE (A&B)
- RELEASE ANCHORS & MANEUVER (WITH ONE END OF CLOTHESLINE TO SOLAR CELL PANELS ON SATELLITE I (A)
- ANCHOR TO STRUCTURE AND ATTACH END OF CLOTHESLINE TO A SUITABLE STRUCTURE (A)
- TRANSFER 6 SOLAR CELL PANELS (ON CLOTHESLINE TO ASTRONAUT A ON SATELLITE (B)
- ASTRONAUT A TO INSTALL 6 SOLAR CELL PANELS ON SATELLITE I
- RELEASE WORKSITE ANCHORS & END OF CLOTHESLINE & MANEUVER TO SOLAR CELL PANELS ON SATELLITE II (A)
- ANCHOR TO WORKSITE & HOOKUP END OF CLOTHESLINE (A)
- TRANSFER 2 SOLAR CELL PANELS (ON CLOTHESLINE) TO ASTRONAUT A ON SATELLITE II (B)
- INSTALL 2 SOLAR CELL PANELS ON SATELLITE II (A)
- TAKE A REST PERIOD (A&B)
- TAKE A REST PERIOD (A&B)
- RELEASE WORKSITE ANCHORS & END OF CLOTHESLINE & MANEUVER TO DIPOLE MODULES ON SATELLITE I (A)
- ATTACH END OF CLOTHESLINE AND WORKSITE ANCHORS (A)
- TRANSFER 4 DIPOLE MODULES TO ASTRONAUT A ON SATELLITE I (B). INSTALL (A)
- MANEUVER TO SATELLITE II & INSTALL 2 DIPOLE MODULES (A)
- ANCHOR TO WORKSITE & HOOKUP END OF CLOTHESLINE
- RELEASE ANCHORS & CLOTHESLINE & MANEUVER TO ACS TANK
- UNSTOW HOSE FOR REPLENISHING FUEL FOR ACS & ATTACH HOSE TO CLOTHESLINE (B)
- TRANSFER HOSE TO ASTRONAUT A (B). ATTACH HOSE TO ACS TANKS (A)
- UNLATCH AND REMOVE HOSE FROM ACS TANK AND ATTACH TO CLOTHESLINE (A)
- MANEUVER TO A SAFE DISTANCE & ANCHOR WHILE TANKS ARE BEING REFUELED (A)
- (IN CM, OPERATING CONTROLS) OPEN VALVE TO START REFUELING (C)
- AFTER COMPLETION OF REFUELING, WILL TURN OF FUEL & DEPRESSURIZE HOSE (C)
- RELEASE ANCHORS (FROM SAFE AREA) AND MANEUVER TO ACS TANK (A)
- ASTRONAUT "A" WILL REMOVE ONE END OF CLOTHESLINE AND WORKSITE ANCHORS, MANEUVER TO S.M. & ANCHOR
- ASTRONAUT "A"&"B" REMOVE END OF CLOTHESLINE (ATTACHED TO S.M.) & STOW IN STOWAGE AREA
- ASTRONAUT "A" TAKE A REST PERIOD
- REPRESSURIZE CM AND REMOVE EVA EQUIPMENT (A&B)
- PERFORM NECESSARY CHECKOUT TO ESTABLISH THAT EXPERIMENT IS PERFORMING PROPERLY
- ENTER THE CM & CLOSE HATCH (A&B)
- ASTRONAUT "A"&"B" RELEASE ANCHORS AT SM AND RETURN TO CM

Notes: 1. This diagram presents an arbitrary sample of the tasks performed to accomplish refurbishment of the antenna assembly.

2. Astronauts' return to the CM at the end of a normal EVA day is omitted to avoid repetition.

Figure 5-1. Flow Diagram for EVA to Extend the Life of the Experiment by Installing New Components and Replenishing the Fuel Supply for the Attitude Control System

Table 5-1. Sample Time Line Analysis

EVENT	ASTRONAUT TASK	ASTRONAUT A	B	C	TIME	CUML.	CUML. EVA A	CUML. EVA B
1	Perform Pre-EVA	X	X		120	120		
2	Cabin Depressurization	X			4	124		
3	Open Hatch and Exit CM	X	X		5	129	5	5
4	Maneuver to Stowage Area, Anchor to Structure, Unstow Clothes Line Rig & Attach it to Structure	X	X		8	137	13	13
5	Release Anchors and Maneuver (with one end of clothes line) to Solar Cell Panels on Satellite No. 1	X			5	142	18	
6	Anchor to a Suitable Structure and Attach Clothes Line to Structure	X			8	150	26	
7	Attach (8) Solar Cell Panels on the Clothes Line ((4) Triangle & (4)Large Panels)		X		5	155	31	18
8	Take a Rest Period	X	X		2	157	33	20
9	Transfer a Large Solar Cell Panel to Astronaut "A" at the Work Site	X	X		5	162	38	25
10	Install one (1) Large Solar Cell Panel on Satellite No. 1	X			3	165	41	
11	Transfer a Triangle Panel to Astronaut "A" at Work Site	X	X		3	168	44	28
12	Install a Triangle Solar Cell Panel on Satellite No. 1	X			3	171	47	
13	Take a Rest Period	X			2	173	49	
14	Release Work Site Anchors and Relocate to Install (4) Additional Panels on Satellite No. 1	X			8	181	57	
15	Repeat Events (9) thru (13) two (2) more times to Install (4) Panels on Satellite No. 1	X			32	213	89	
16	Release Work Site Anchors and One End of Clothes Line and Maneuver to Panels on Satellite No. 2	X			8	221	97	
17	Anchor to a Suitable Structure on Work Site and Attach Clothes Line to Structure	X			7	228	104	
18	Repeat Events (9) thru (13) one time to Install (2) Solar Cell Panels on Satellite No. 2	X			16	244	120	
19	Release Work Site Anchors, Remove Clothes Line Rig, Return to Stowage Area and Anchor	X			10	254	130	
20	Take a Rest Period	X			2	256	132	
21	Remove Clothes Line Rig which is attached to SM Structure and Stow in Stowage Area	X	X		4	260	136	32
22	Release Anchors at SM and Return to the CM	X	X		5	265	141	37
23	Enter the CM and Close Hatch	X	X		10	275	151	47
24	Repressurize the CM and Remove the EVA Equipment	X	X		65	340		
						END OF DAY 1		
25	Perform Pre-EVA	X	X		120	460		
26	Cabin Depressurization	X			4	464		
27	Open Hatch and Exit CM	X	X		10	474	10	10
28	Maneuver to Stowage Area, Anchor to Structure, Unstow Clothes Line Rig and Attach One End to Structure	X	X		8	482	18	18

Table 5-1. Sample Time Line Analysis, Contd

EVENT	ASTRONAUT TASK	ASTRONAUT A	B	C	TIME	CUML.	CUML. EVA A	CUML. EVA B
29	Release Anchors and Maneuver (with one end of clothes line) to Dipole Modules on Satellite No. 1	X			5	487	23	
30	Anchor to a Suitable Structure and Attach Clothes Line to Structure	X			7	494	30	
31	Attach (6) Dipole Modules on the Clothes Line		X		5	499	35	23
32	Transfer a Dipole Module to Astronaut "A" at Work Site	X	X		5	504	40	28
33	Flip Expended Module to Stowage Position and Install New Module for Satellite No. 1	X			3	507	43	
34	Take a Rest Period	X	X		2	509	45	30
35	Repeat Events (32) thru (34) three (3) More Times for Satellite No. 1	X	X		30	539	75	51
36	Release Work Site Anchors and Clothes Line and Maneuver to Dipole Modules on Satellite No. 2	X			10	549	85	
37	Anchor to a Suitable Structure on the Work Site and Attach Clothes Line to Structure	X			6	555	91	
38	Repeat Events (32) thru (34) two (2) More Times for Satellite No. 2	X	X		20	575	111	65
39	Release Work Site Anchors and Clothes Line, Maneuver to SM and Anchor	X			10	585	121	
40	Remove Clothes Line Rig (Attached to SM Structure) and Stow in Stowage Area	X	X		4	589	125	69
41	Release Anchors at SM and Return to the CM	X	X		5	594	130	74
42	Enter the CM and Close Hatch	X	X		10	604	140	84
43	Repressurize the CM and Remove the EVA Equipment	X	X		65	669		
						END OF DAY 2		
44	Perform Pre-EVA	X	X		120	789		
45	Cabin Depressurization	X			4	793		
46	Open Hatch and Exit CM	X	X		10	803	10	10
47	Maneuver to the SM and Anchor to the Structure	X	X		7	810	17	17
48	Unstow Clothes Line Rig and Attach One End to Structure on SM	X	X		4	814	21	21
49	Release Anchors and Maneuver (with one end of clothes line) to an ACS Tank	X			4	818	25	
50	Anchor to the Work Site and Attach Clothes Line to Structure	X			4	822	29	
51	Unstow Fuel Hose for ACS Tanks From Stowage Area (concurrent with Event 50)		X					
52	Attach Hose Nozzle to Clothes Line and Transfer to Astronaut "A" at Work Site		X		4	826	33	25
53	Take a Rest Period	X	X		2	828	35	27
54	Remove Fuel Hose Nozzle from Clothes Line and Attach to ACS Tank Using Toggle Latch	X			4	832	39	
55	Remove Work Site Anchors, Maneuver to a Safe Distance and Anchor	X			5	837	44	

Table 5-1. Sample Time Line Analysis, Contd

EVENT	ASTRONAUT TASK	ASTRONAUT A	B	C	TIME	CUML.	CUML. EVA A	CUML. EVA B
56	Open Valve in CM to Start Refueling			X				
57	After Completion of Refueling, Turn Off Fuel and Depressurize Hose (From CM)			X				
58	Remove Anchors (from safe area) and Maneuver to ACS Tank (just refueled)	X			4	841	48	
59	Unlatch and Remove Hose from ACS Tank and Attach to Clothes Line	X			4	845	52	
60	Take a Rest Period	X			2	847	54	
61	Remove one end of Clothes Line (with hose attached), Release Work Site Anchors & Maneuver to Next ACS Tank	X			6	853	60	
62	Anchor to Work Site and Attach Clothes Line to Structure	X			4	857	64	
63	Repeat Events (54) thru (60) (5) times, and Events (61) thru (62) (4) Times for Remaining (5) ACS Tanks	X		X	140	997	204	
64	Remove one end of Clothes Line (with hose attached) and Remove Work Site Anchors	X			4	1001	208	
65	Maneuver to SM and Anchor to Structure	X			4	1005	212	
66	Remove Clothes Line Rig (Attached to SM Structure) and Stow in Stowage Area	X	X		4	1009	216	31
67	Release Anchors at SM and Return to the CM	X	X		5	1014	221	36
68	Enter the Cm and Close Hatch	X	X		8	1022	229	44
69	Repressurize CM and Remove EVA Equipment	X	X		65	1087		
70	Perform Necessary Checkout to Establish Experiment is Performing Properly	X	X	X	50	1137		

END OF MISSION

Astronaut A and his backup man B have visual as well as radio contact in order that B may assist A in an emergency. The crewman remaining in the command module (operating controls) is Astronaut C.

5.4 ASTRONAUT SUPPORT EQUIPMENT REQUIREMENTS. The following equipment is required to accomplish the repair and replacement of components of the crossed-H interferometer radio astronomy antenna.

a. An Apollo Block II space suit.

b. A portable life support system.

c. Fixed and portable work site anchors including dutch shoes (The dutch shoes will not be part of the structure).

d. A communication system.

e. Fixed or portable illumination system.

f. A hand held maneuvering unit.

g. A 50-ft tether.

h. Spares

A portable life support system and the space suit will provide life support, biomedical data, and communications. An umbilical for EVA was considered desirable but the Apollo CM cannot presently support long time or distant EVA without a portable life support system. The study of modifications to the CM that will allow the use of an umbilical for EVA in lieu of a PLSS is recommended. The umbilical has the advantages of being lighter, more mobile (reducing fatigue), easier to attach, allows the astronaut to work in smaller areas and does not require refilling. The hand held maneuvering unit could be available to the astronaut as a safety factor in the event of an emergency.

The astronaut carries his dutch shoes and waist restraints to use as work site anchors. Small equipment may be stowed in the CM, but larger parts would have to be stowed in Sector IV of the Service Module.

The communication system is required to provide vocal contact between the EV crewman, his parent spacecraft, and any other crewman that may be required for assistance.

In the process of repair and replacement of components, work site lighting is required for dockside operations and for shadows cast by antenna components. The helmet-mounted lights are preferred to other portable systems to minimize tool or limb interference with illumination of the task.

A power supply system is required for the astronaut's equipment, illumination system, telemetry, voice communication, environmental control, etc. This power supply should be so designed to preclude periodically returning to the parent spacecraft for recharging during the allotted EVA time.

SECTION 6

RELIABILITY

6.1 SCIENTIFIC MISSION RELIABILITY. As defined by NASA, the crossed-H inter-
ferometer has three flight objectives previously discussed in the introduction. By
definition, the crossed-H interferometer is a manned space flight payload. It is of
interest, however, to perform a reliability analysis of the antenna, considering it as
an unmanned radio astonomy satellite as a comparison with the manned concept. The
first two flight objectives could not be fulfilled by flying the antenna in an unmanned
mode, however, the radio astonomy mission could still, to a large degree, be accom-
plished. The results of a very brief analysis are presented herein.

6.1.1 Reliability in Design. To provide an acceptable value for probability of mis-
sion success, the satellites for the interferometer experiment have been designed for
unattended operating periods of up to one year. The reliability of the two satellites
operating together must be greater than that of a single present-day communications
satellite. Although it is possible to design the satellites with sufficient maintainability
by EVA to effect restoration from most failure modes, it is anticipated that manned
excursions to them will be infrequent. Therefore, if system failures occur too fre-
quently, completion of the planned experiment would take years, with the satellite sys-
tem in an inoperative state over most of the period. Also, certain failure modes pre-
clude any possibility of system restoration, regardless of maintainability features.

It is assumed that parts and components used for these satellites will be within the
state of the art for the 1968-1972 period. No large-scale component development for
this program is anticipated. The necessary reliability will be achieved primarily by
carefully minimizing the probability of initial failure modes and the correction of un-
avoidable failures by EVA.

There is a class of failure modes affecting the operation of the attitude control system
(ACS) which is most critical for the mission. Any mode that causes uncontrollable
thrusting from one or more reaction jets results in the loss of equilibrium of the affec-
ted satellite. Such failure modes include open failure of the thruster valve or a spuri-
ous command to open from the jet controller. The command could be generated in the
ACS electronic control subsystem or in the navigation system. Also, an electrical
power failure during an attitude change would result in continued motion of the satellite.

It is conceivable that such motion would prevent EVA repairs. It is improbable that
an astronaut could safely get close enough to effect a repair, unless the satellite could
be stabilized by remote control. However, stabilization could prove difficult, due to
wrapping or twisting of the tether with possible tether damage.

There is a method whereby such mission failure can probably be prevented. It would include an electronic circuit that would sense the failure of attitude control, compute the compensating command, if any, and at least override the failed jet with selective shutoff. Such a system backed up by a minimal redundancy of jets should receive further consideration.

Part and component redundancies are being employed in design to enhance probability of mission success. A certain amount of redundancy is inherent in the tether and its two reels, one per satellite, and among the attitude control thrusters. Solar arrays are designed for over-capacity, so that the failure of a few cells over a period of time can be tolerated. Standby drive motors are provided for the booms and dipole antennas. It is certain that the detailed design of electronic subsystems, including instrumentation, communications, navigation, and control, will necessarily include redundancy at some level. A sufficient variety of integrated circuits will probably be available so that they can be used in most applications in these satellites. Not only does their small size and low cost make redundancy attractive, but also single-chip reliability is continuously increasing as experience is gained in quality improvement.

Finally, loose operating tolerances are employed whenever feasible. In this manner the system remains operable in spite of normal drift of part values. A commonly employed example of the use of loose tolerances for increased reliability and performance is a digital information system. This kind of system is relatively immune to changes in original strength, gain or noise levels. With such features, the communications system, for example, can be initially adjusted for maximum sensitivity and output. Reasonable changes in part values over the unattended period of operation may degrade performance somewhat, but the necessary data and commands can still be interchanged.

6.1.2 Unmanned System Reliability. An initial reliability prediction was performed for a completely unattended pair of satellites, so that the effects of astronaut participation on probability of mission success discussed in the following subsection can be evaluated. The prediction is for both spacecraft operating together and includes the effects of launch into orbit. It does not include any ground systems or the booster and upper stage required to place them into orbit. The prediction assumes a completely unmanned space mission. The results are summarized by subsystem and by mission phase in Table 6-1. Similar subsystems in each spacecraft are combined into a single subsystem for the entire satellite structure. Structural components, including the tether, are not included, since sufficient margins of safety are incorporated to assure their integrity for the duration of the mission. Component failure rates used are mostly based on those in the Failure Rate Data Handbook (FARADA), published by the U.S. Navy, FMSAEG, Corona, California. For the electrical power subsystem data were extracted from a paper, An Evaluation and Comparison of Power Systems for Long Duration Manned Space Vehicles, by J. G. Krisilas and H. J. Killian of the Aerospace Corporation, presented at the Intersociety Energy Conversion Engineering Conference of AIAA, ASME, IEEE, and AICE, 26 to 28 September 1966.

Table 6-1. Crossed-H Interferometer Cumulative Mission Reliability

SUBSYSTEM	DEPLOYMENT	14 DAYS	1 YEAR	428 DAYS	2 YEARS
Power System	0.999989	0.99984	0.9952	0.9937	0.986
Radiometer Equipment	~1.0	0.999990	0.99906	0.9986	0.9964
Interferometer Structure and Mechanical Drives	0.99937	0.9911	0.817	0.787	0.658
Instrumentation and TLM	~1.0	0.999987	0.9918	0.9988	0.9968
Initial Deployment	0.99957	0.99957	0.99957	0.99957	0.99957
Navigation and Attitude Control Systems	0.999995	0.99989	0.9932	0.9906	0.979
Command and Control Systems	~1.0	0.999980	0.9967	0.9950	0.987
Total Assembly	0.9989	0.9904	0.803	0.768	0.622

The failure rates were adjusted to ground laboratory conditions when required, to facilitate applying them to orbital environment. The ground rates were divided by two as a conservative allowance for improvement in state-of-the-art reliability. For the orbital launch phase, the adjusted ground rates were multiplied by 80, in accordance with MIL-STD-756A.

Most electrical and electronic components were assumed to contain part and circuit redundancy, the effect of which is equivalent to redundancy at the component level. Also, redundancies presently incorporated in design are included. The result of actual and assumed redundancy on the analysis is an "aging" effect, due to the mixed exponential distribution of time between failures of the system. The effect simulates not only redundancy, but system degradation due to drift of operating parameters. A duty cycle of 10% is assumed for the drive chain for the booms, antennas, and tether reels. This amount exceeds by far the actual anticipated duty cycle, so as to include effects of formancy and start-up.

The prediction indicates that the major portion of system unreliability is in the interferometer subsystem, including the drive mechanisms. It appears that the six drive transmissions for the booms and dipoles are together the major contributors to unreliability. This fact is not due to any higher failure rates being attributed to the transmissions. Rather, their relatively high unreliability is due to assuming no redundancy in these components. It is difficult to provide redundant transmissions without adding appreciable weight and volume requirements. Since failure rate data for the specific components to be used are not now available, conservative rates based on FARADA were used. Unmanned mission reliability will be greatly improved, simply by judicious selection of transmissions, if the principles of part and component redundancy and loose tolerances where possible are employed as assumed. Furthermore, the use of man during checkout in orbit and during annual system renovation brings the probability of mission success to an acceptable level.

6.2 MAN'S IMPACT ON MISSION RELIABILITY. By diligent application of design reliability practices the unmanned reliability of a satellite system could attain virtually any selected goal. However, the cost, development schedule, size and weight might well become prohibitive. Using man during system checkout and deployment and during scheduled or unscheduled periods of repair and renovation makes possible the achievement of virtually any selected goal for probability of mission success. Not only can the man replace failed components, but he can make the adjustments to restore the system to a peak operating condition and replace components and expendables likely to give out in the near future. Man can also be used to uprate the scientific quality of the antenna through subsequent rendezvous and updating the receiving systems with new, more sophisticated electronic modules, if desired.

Design criteria for the interferometer satellite include features for maintainability in an EVA environment, such as modules for replaceable units, large knobs and handles for making adjustments and clamping, and handholds and belt attachments for restraining

the astronaut. The panels of the center body can be opened for ready access to any component within. Provisions for ready accessibility to components are also provided in the dipole drive assemblies. Safety provisions include flexible antenna elements and soft coatings over sharp corners to avoid snagging or tearing a pressure suit or astronaut tether.

Table 6-2 shows the contribution of man to the system reliability at various times in orbit, including manned excursions at one year and at two years of orbital life of the antenna assembly. Since the design is not sufficiently detailed to determine exactly which component failures can be corrected by maintenance, it is assumed that the astronaut will repair 90% of the failures. Such repairs include actions to restore the system to its peak operating condition, whether or not complete system failure has occurred. Failures resulting in loss of spacecraft attitude stability preclude possibility of repair, unless stability can first be restored by remote control. These failure modes have been mentioned in subsection 6.1.1.

The probability values stated in Table 6-2 are simply the probabilities that the subsystems and systems are operable at the given points in time. After the assembly of the antenna system it is assumed that the crew remains with the assembly for 14 days to check it out and perform any necessary repairs and adjustments. Meantime, to restore the system to operation, should be less than 5 hours, so that 14 days is sufficient time to effect virtually all repairs that are possible. The satellite assembly is assumed to start its mission at the start of checkout. After the 14-day checkout, it continues for 50 weeks unattended. The annual renovation period then starts, and a crew is available also to restore the system to peak operating condition, in addition to replacing items of limited life. Renovation, repairs and checkout are assumed to be completed in 14 days. This crew visit begins the second annual cycle of the system, similar to the first, with the additional task of system renovation.

Of interest is the amount that the use of man enhances the probability of mission success. For a 428-day mission without astronaut participation, the probability of mission success is estimated to be 0.768, from Table 6-1. If man participates in the initial checkout period and in the first annual renovation, at the end of the first year, as described above, the antenna assembly must operate unattended for two periods totalling 400 days. If it is restored from a failed state after 351 days of successful operation, it must operate successfully for the balance of 49 days or a second renovation and repair must be performed. However, from Table 6-1, it appears that the probability of success is improved to 0.95. This is discussed in further detail.

The probability of operation without failure during the unattended periods is $(P_M)^2$ R_{400}, where P_M is the probability of successful operation at the completion of a manned period and R_{400} is the conditional probability of successful unattended operation for 400 days, given successful operation after each manned period. R_{400} was approximated by taking R_{351}, the conditional reliability for 351 days of unattended

Table 6-2. Probability of Operation, Manned Checkout and Renovation Antenna Assembly

SUBSYSTEM	14 DAYS	1 YEAR	1 YEAR + 14 DAYS*	2 YEARS	2 YEARS + 14 DAYS*
Power	0.999984 / 0.00014	0.9954 / 0.0002	0.99959 / 0.0044	0.9950 / 0.009	0.99918 / 0.013
Radiometer Equipment	~1.0 / 0.000010	0.99907 / 0.00001	0.999979 / 0.00092	0.99905 / 0.0026	0.999955 / 0.00036
Interferometer Structure and Mechanical Drives	0.99911 / 0.0080	0.823 / 0.006	0.978 / 0.162	0.806 / 0.148	0.958 / 0.302
Instrumentation	~1.0 / 0.000013	0.99930 / 0.00012	0.999976 / 0.00080	0.99917 / 0.0024	0.999955 / 0.0032
Deployment	~1.0 / 0.00043	~1.0 / 0.00043	~1.0 / 0.00043	~1.0 / 0.00043	~1.0 / 0.00043
Navigation and Attitude Control	0.999992 / 0.00010	0.9932 / 0.0000	0.99976 / 0.0066	0.9930 / 0.0024	0.99947 / 0.00010
Command and Control	~1.0 / 0.000020	0.9967 / 0.0000	0.999944 / 0.0032	0.9966 / 0.0016	0.99985 / 0.003
Total Assembly	0.999087 / 0.0087	0.810 / 0.007	0.977 / 0.174	0.792 / 0.170	0.956 / 0.334

Note: Upper figure is probability; lower figure is improvement due to man. The figures can be applied only to the third flight objective of the structure (scientific mission). By definition, the structure requires man to accomplish the other two objectives.

*Improvement over 1 and 2 years unmanned operation, respectively.

operation, setting it equal to $e^{-x}351$ and solving for x; R_{400} was determined by raising e to the $\dfrac{-400x\,351}{351}$ power. Due to redundancy in the system, x does not actually vary linearly with time, but increases at a slightly faster rate. However, the resulting estimate is slightly conservative, considering that the system is at least partially renewed at the end of a year. This conservatism allows for aging of non-replaced components. Thus one possible condition for mission success is considered.

The other possibility of mission success is that of successful completion of checkout, failure any time during the next 351 days, successful restoration and completion of the balance of the 400 days of unattended operation. The possibility of a second renovation mission by the crew is neglected, even if the balance of the unattended 400 days takes the mission duration into the third year. Letting the failure density during unattended operation by $f(t_1)$, the probability of failure during the first 351 days of unattended operation is

$$1 - R_{351} = \int_0^{351} f(t_1)\, dt_1$$

The reliability for the balance of the 400 days of unattended operation is given by

$$R_{400-t_1} = \int_{400-t_1}^{0} f(t)\, dt$$

The joint probability of failure during the first 351 days and success during the remainder of the 400 days is

$$\int_0^{351} t(t_1) \int_{400-t_1}^{\infty} f(t)\, dt\, dt_1$$

Using the previous assumption that $R = e^{-x}$, where x varies approximately linearly with time, x can be set equal to λt. The failure density $f(t_1)$ is then $\lambda e^{-\lambda t_1}$. Substituting that density function in the double integral above and integrating gives:

$$(1 - R_{351})\, R_{400-t_1} = 351 \lambda e^{-400\lambda} = x_{351} R_{400}$$

The probability of successful checkout is P_M, as before; the probability of successful restoration and checkout during renovation is $0.9\, P_M$, because of the 90% effectiveness assumed for repair.

Adding the probabilities of the two conditions for mission success, allowing just one renovation mission, gives the total probability of mission success:

6-7

$$P_s = (P_M)^2 R_{400} + 0.9 (P_M)^2 x_{351} R_{400} = (P_M)^2 R_{400} (1 + 0.9 x_{351})$$

$$= (0.999087)^2 \, 0.787369 \, [1 + 0.9 \, (0.20977)] = 0.934$$

If a single renovation mission could be dispatched after a system failure, even if the failure occurs beyond the end of the first year, it can be shown approximately that

$$P_s = (P_M)^2 R_{400} \left(1 + \frac{360 \, x_{351}}{351} \right) = 0.953$$

Allowing more than a single renovation mission raises the probability of mission success for two reasons:

a. More time in manned operation is available; the probability of keeping the system in operation during manned periods is appreciably greater than during unmanned periods of equal length.

b. More chances are available for system restoration in the event of more than one failure.

SECTION 7

RESEARCH, DEVELOPMENT, TEST AND ENGINEERING

7.1 INTRODUCTION. This chapter is the RDT&E plan for the crossed-H interferometer, documenting the Task 5 approved study work statement items.

The purpose of the RDT&E plan is to provide a focal point for the planning documents and information necessary to define all steps required to achieve a functioning orbital antenna as part of the AAP. Definition is required in sufficient detail to support NASA resource allocations between the various alternatives to identify requirements for manpower, research development and test facilities, and to define schedule interactions and budgetary planning data to achieve, for AAP, maximum utilization of resources.

The RDT&E plan provides:

a. Work breakdown structure.

b. Prerequisite orbital experiments.

c. Research, manufacturing, test, and support plans.

d. Schedule.

e. Cost analysis.

The work breakdown structure incorporates the system elements and identifies the tasks associated with each. Based on guidelines established in the preceding design chapters, the supplementary research, manufacturing, test and support plans of this chapter, the schedule and cost data (keyed to the work breakdown structure) have been generated.

The level of definition throughout the RDT&E plan is selected to be commensurate with the planning indicated in the design chapters, and tailored to subsequent completion of NASA Form 1346 for the crossed-H interferometer experiment.

The system elements of the crossed-H interferometer are shown in the system block diagram, Figure 7-1. Description of the hardware items is found in the preceding design sections. A plan specifically for facilities has not been incorporated in this RDT&E discussion, but where requirements for facilities are recognized, they are included in the manufacturing or test plan as appropriate. Discussion of personnel and data are included in the support plan.
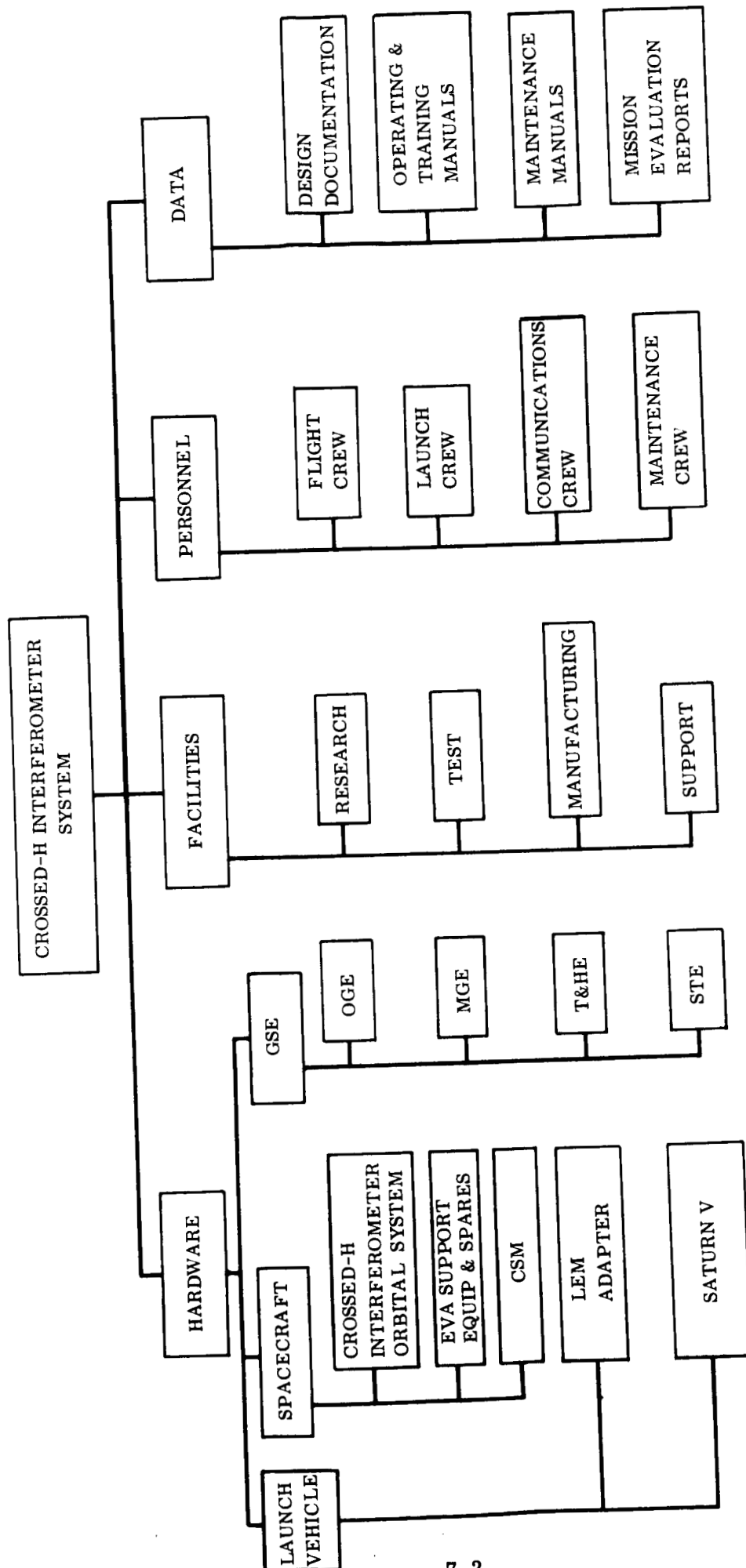
Figure 7-1. Crossed-H Interferometer System Block Diagram

7.2 <u>WORK BREAKDOWN STRUCTURE.</u> A work breakdown structure is shown in Figure 7-2. Development of the experiment is assumed to be within the Apollo Applications Program so that the other major segments required for flight are GFE to the experiment project. Incorporation of other experiments into the same launch vehicle, while shown as potentially feasible, is also assumed to be GFE. In addition to basic hardware the following elements are shown:

a. Program Management. This element includes all of the contractor's program office activities in general categories of technical management, control, and documentation.

b. Systems Engineering. Two work areas are defined - contractor control and subcontractor control. In the first, systems engineering performs technical analyses and initiates design requirements; reliability standards are established; and verification of design is achieved through an integrated test plan. Subcontractor control is achieved through selection, surveillance and product inspection.

c. Support Operations. This element includes those ancillary operations necessary for the accomplishment of the mission - training of personnel, preparation of procedures, participation in prelaunch and launch activities.

7.2.1 <u>Aerospace Equipment.</u> The vehicleborne equipment is shown under two different classifications. First, the experiment equipment is shown with associated design, fabrication, and assembly tasks. The interferometer, adjustments, and associated components are included in this group. Second, the spacecraft equipment is similarly defined. This equipment is used in support of the antenna and typically includes telemetry, tracking, command, and power subsystems.

7.2.2 <u>Ground Support Equipment.</u> Group support equipment (GSE) typically includes checkout equipment, operational equipment, handling equipment, and other special equipment. To the greatest extent practicable, fixtures will be used both for tooling/assembly and test.

7.2.3 <u>Facilities.</u> While the four standard types of facilities are listed, only one new facility item is currently identifiable: It is the zero-g deployment facility. Deployment of each of the telescoping truss-work booms in the horizontal position will require a test-bed with an overhead crane system to which the assembly is attached by negator springs. Alternatively, the deployment will be accomplished upward in a vertical tower with pulley and track systems to accommodate the appropriate counterweights.

7.3 <u>PREREQUISITE ORBITAL EXPERIMENTS.</u> The following prerequisite orbital experiments will, if conducted in space in support of the large space structures development program, increase the probability of successful accomplishment of the crossed-H interferometer.

Figure 7-2. Work Breakdown Structure

AAP FLIGHT 5--

- LEM ADAPTER SYSTEM *
- SATURN V LAUNCH VEHICLE SYSTEM *
- OTHER AAP EXPERIMENT SYSTEMS
- CROSSED-H INTERFEROMETER
- APOLLO CSM SYSTEM *
- ORBITAL SUPPORT OPERATIONS *

CROSSED-H INTERFEROMETER / PRINCIPAL INVESTIGATOR

PROGRAM MANAGEMENT
- TECHNICAL MANAGEMENT
  - PLANNING
  - COORDINATION
  - REVIEWS
- CONFIGURATION CONTROL
  - ENGR. RELEASE
  - CHANGE CONTROL
  - SUBCONTRACTOR CONTROL
- SCHEDULE CONTROL
- FINANCIAL CONTROL
  - SALES ORDERS
  - WORK ORDERS
  - WORK ASSIGNMENT PLANS
  - SHOP TRAVELERS
- DOCUMENTATION
  - REPORTS
  - PRESENTATIONS
  - PROGRAM PLANS
  - DESIGN DATA PACKAGE
  - COMPUTER PROGRAMS

SYSTEMS ENGINEERING
- TECHNICAL ANALYSIS
  - DESIGN ANALYSIS
  - LOADS
  - STRESS
  - DYNAMICS
  - THERMAL
  - COMPUTER PROGRAMS
- TESTING
  - ACCEPTANCE TESTS
  - STRUCTURAL TESTS
  - ENGR. MODEL TESTS
  - INTEGRATED SYSTEMS TESTS
- RELIABILITY & QA
  - REL. APPORTIONMENT
  - FAILURE MODE ANALYSIS
  - FABRICATION INSPECT.
  - ASSEMBLY INSPECT.
  - TEST INSPECT.
  - DESIGN REVIEW
  - CALIBRATION
- SUBCONTRACTOR CONTROL

EXPERIMENT DESIGN, FABRICATION, & TEST ①
- DESIGN
- FABRICATION ASSEMBLY
  - ANTENNA SUBSYSTEM
    - TELESCOPING BOOM (2)
    - BOOM DEPLOYMENT MECHANISM
    - DIPOLE STEM ASSY
    - DIPOLE HEAD ASSY (2)
    - STEM DEPLOYMENT MECHANISM
    - TAPE TETHER
    - TETHER DEPLOYMENT MECHANISM
  - RADIO METER SUBSYSTEM
    - RECEIVER
    - SIG GENERATOR
    - STANDARD LOAD
  - POINTING SUBSYSTEM

SUPPORT OPERATIONS
- TRAINING OPERATIONS
  - FLIGHT CREW (EVA OPERATIONS) (ZERO-G SIMULATIONS)
  - LAUNCH CREW
  - ORBITAL SUPPORT CREW (COMMUNICATIONS) (COMPUTER OPERATIONS)
- LAUNCH SUPPORT OPERATIONS
  - PRELAUNCH (DOCUMENTATION) (ASSEMBLY & C/O) (COUNTDOWN)
  - LAUNCH

SPACECRAFT DESIGN, FABRICATION & TEST ①
- DESIGN
- FABRICATION ASSEMBLY
  - POWER SUBSYSTEM
    - SOLAR ARRAY
    - BATTERY
    - POWER CONDITIONING
    - HARNESS & CABLING
  - TELEMETRY TRACKING & CONTROL SUBSYSTEM
    - TRANSMITTERS (2)
    - MULTIPLEXER/ENCODER
    - SIGNAL CONDITIONER
    - TIME CORRELATION
    - COMMAND DECODER
    - AUDIO CENTER
    - ANTENNAS
    - RECEIVER ②
  - ATTITUDE CONTROL SUBSYSTEM
    - ATTITUDE SENSORS
    - REACTION SYSTEM ASSY
    - LINKAGE
    - COMPUTER & CONTROL ELECTRONICS
  - STRUCTURES & SUPPORTS SUBSYSTEM

GROUND SUPPORT EQUIPMENT
- CHECKOUT EQUIPMENT
  - EXPERIMENT C/O CONSOLE
  - ANTENNA PATTERN CONSOLE
  - SPACECRAFT C/O CONSOLE
- OPERATIONAL EQUIPMENT
  - LAUNCH CONSOLE
  - COMMAND CONSOLE
  - DATA STATION (RECEIVER) (DECOMMUTATOR) (RECORDER) (COMPUTER) (PLOTTER)
- HANDLING EQUIPMENT
  - PALLET
  - PROTECTIVE COVER
  - SHIPPING CONTAINER
  - HANDLING CART
  - HOISTING SLINGS
- SPECIAL TOOLING
- TEST FIXTURES
  - ASSEMBLY FIXTURE
  - ZERO-G DEPLOYMENT FIXTURE

FACILITIES
- RESEARCH
- MANUFACTURING
- TEST
- OPERATIONS

NOTES ~1. * GFE TO THE CROSSED-H PROJECT
① FOR EACH OF TWO 2 SATELLITES (UPPER & LOWER).
② RECEIVERS - 1 EA. UPPER 2 EA. LOWER

7-4-1

In addition to the desirable orbital experiments listed below, two areas requiring technological advancement are of concern to this study, but may be satisfied by related satellite programs. The dynamic behavior of tethered, gravity gradient stabilized satellites in synchronous orbit could be verified by other systems placed in orbit prior to the crossed-H interferometer, such as the tethered orbiting interferometer (TOI) studied by the Applied Physics Laboratory and R. Stone of GSFC. A second area of concern is the dynamic/thermal behavior of long extendible tubular elements (for dipole application in the case of the crossed-H, Section 7.3.3). Here again, GSFC is conducting tests of benefit to the proposed crossed-H interferometer.

7.3.1 Clothesline Supply.

Purpose: To demonstrate the practicability of supplying a remotely positioned EVA astronaut with tools and materials and for the return of these as required from a parent spacecraft through a "loop clothesline" EVA aid.

Equipment Required: Basic hardware is a loop of line under tension between two pulleys, and a traveller to which items are fastened for transport.

Procedure: An EVA astronaut will move to a remote position on the large space structure, carrying with him the clothesline assembly attached to a tether and to the parent spacecraft. A second astronaut will monitor the paying out of the tether from a small power driven spool. When one pulley is fastened to an appropriate remote location on the structure, the second astronaut will retract the tether and pull in the second pulley. The loop of line feeds out from a spring-loaded take-up spool in the traveller. The clothesline can then be used for the transfer of tools and materials.

Measurements: The primary measurement is the subjective evaluation of the clothesline supply as an EVA aid. Some of the parameters which can be measured, adjusted and effect noted are:

a. Tension in the loop, strain on the structure.

b. Degree of control by the operating astronaut over the materials transferred.

c. Effect of damping in the take-up spool.

d. Effect of ACS maneuvering on the clothesline while materials are being transferred.

e. Time to install clothesline, reposition, and return to storage.

7.3.2 Astronaut Locomotion Loads.

Purpose: To determine actual loads on the typical large space structures resulting from astronaut locomotion in contact with the structure while in the space environment. The results will be used to generate realistic structural design loads for the crossed-H interferometer structures.

Equipment: A typical structural assembly is used in conjunction with the parent spacecraft which provides the necessary mass for reacting loads against, and the ACS capability to generate loads in the structure while the astronaut is performing EVA on the structure. The structure will be an open framework of various length and cross-section struts, typical hand-holds and tether attachment points, and the necessary instrumentation.

Procedure: The structural assembly is erected on the parent spacecraft, probably with EVA assistance, and instrumented. The astronaut then moves about the structure, possibly performing other experiments (e.g., that of Section 7.3.1).

Measurements: The following types of measurements are made:

a. Deflections.

b. Strains.

c. Shock loading.

d. Accelerations.

e. Temperatures.

f. Resonant frequencies and damping characteristics.

7.3.3 Boom Deflection.

Purpose: To determine the static and dynamic characteristics of the dipole booms in the space environment.

Materials: Deployable tubular boom antennas (bi-metallic screen, plated and non-plated conventional stem, and perforated booms) and instrumentation.

Approach: The booms are deployed, then held in various attitudes with respect to the earth and sun while measurements are made. The booms are torqued and flexed, using the parent spacecraft ACS.

Measurements: Measurements made on the deployed booms are:

a. Tip deflection, radius of curvature, and skewness.

b. Stiffness, natural frequency, and damping coefficients.

c. Dipole characteristics.

d. Temperatures.

7.4 <u>RESEARCH PLAN</u>. Design of the crossed-H interferometer is based on current state of the art in development of materials and construction details. Development of the large space structures required will be primarily concerned with testing, refining manufacturing techniques, and providing support necessary for the large structures. While not specifically required, research in the following areas will contribute to performance, cost or schedule improvements:

a. Instrumentation for measuring boom deflection in space.

b. Analytical modeling and analyses of effects of distortion on the large dipoles.

c. Techniques to reduce manufacturing tolerances on screen-boom tubing.

d. Dynamic analytical models of complex space structures.

e. Control system simulation for large, flexible space structures.

f. Evaluation of astronaut EVA performance, capabilities, and developments, and predictions for future capabilities.

7.5 <u>MANUFACTURING PLAN</u>. Manufacturing processes required to fabricate components of the crossed-H interferometer are essentially those processes and techniques familiar to airframe and aerospace manufacturers and utilized extensively by General Dynamics.

The 7000-series aluminum selected for general structural use presents no unusual problems as to machining, fabrication or assembly. Metal bonding of the solar panel structure will afford an uninterrupted surface for application of solar cells and this is a technique currently in use in many metal-joining applications.

A "pilot line" concept applied to manufacturing the crossed-H interferometer will utilize some sub-assembly positions. Assemblies will be complete and flow to the final assembly position when the complexity, manufacturing processes, or test requirements indicate this to be advantageous.

The crossed-H interferometer manufacturing sequence (Figure 7-3) depicts an appropriate manufacturing breakdown and sequence of manufacturing events. Some events, such as center-body assembly, boom final assembly, and checkout and test, are performed in the same manufacturing position and area without assembly movement. Other events will, of course, occur in various manufacturing areas due to the difference in the nature of the manufacturing processes involved.

7.5.1 <u>Final Assembly</u>. Upon completion of the upper and lower center-body structure assemblies, the structures are mated (interconnect structure) and supported on a fixture that will be used during experiment final assembly, checkout and shipping. Consideration of the use of the actual launch support for this fixture would eliminate transfer of the assembly later. Mechanical and electrical components as depicted in the manufacturing sequence are installed.
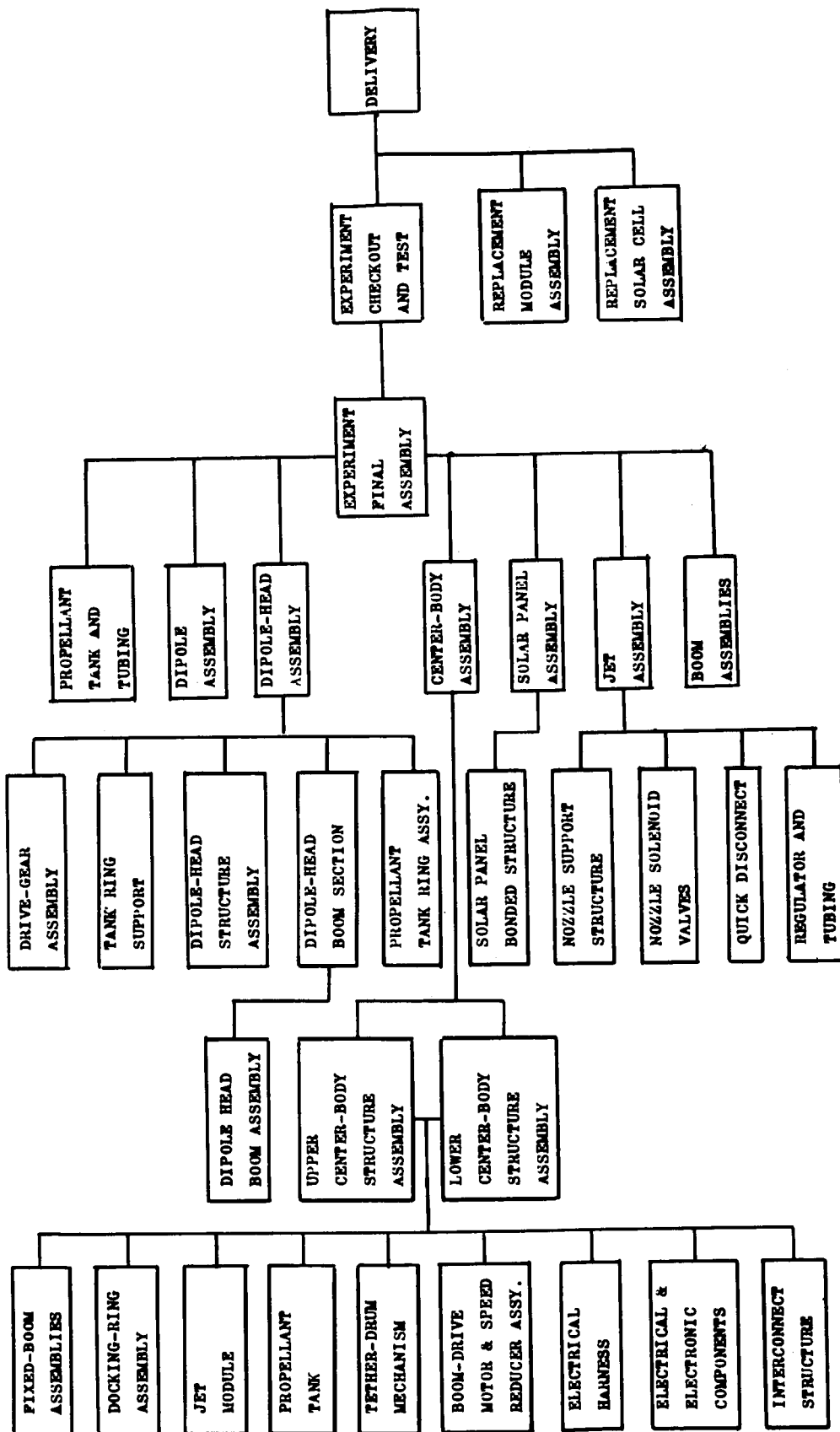
Figure 7-3. Manufacturing Sequence

After installation of the first section of movable boom at the four boom positions, an extension and retraction cycle is performed. Addition of the remaining boom sections and the dipole head assembly may then be accomplished.

Boom assembly and test requires special attention to fabrication and assembly methods. Precise control of critical interface areas is required to assure frictionless and unhampered movement during extension and retraction. This is best accomplished by constructing all boom section assemblies in a single special fixture having the stability and tolerances of a "tooling dock". Reference points are established to enable adjustment of rollers and gears so that only minor adjustment, if any, is needed at final assembly.

Three boom caps are received completely machined and located in the assembly fixture holding the control surfaces interfacing with subsequent assemblies. The fixture is adjustable to the size of the different boom sections by close tolerance bored pin positions. In this manner the fixture functions both as an assembly tool and a gage. The completely assembled boom is extended and retracted while supported on an air-bearing table. Booms are operated singly by repositioning the center-body assembly support as required.

Hinged solar panels, jet assembly dipole assembly, and remaining components are now installed. Clearance around supporting structure is provided to permit partial simultaneous extension of dipoles to verify operation. Final steps of antenna checkout and testing are completed at this time.

7.5.2 Major Assembly and Test Fixture. This fixture supports the mated center-bodies for final assembly and test. It provides an air-bearing table to support boom sections during extension and retraction tests. The tool requires a floor area approximately 15 ft by 50 ft and provides for extension of a single boom at one time to its full length. The center-body support structure of the tool may be repositioned and rotated to enable testing of all four booms. Fixture is located with sufficient vertical clearance to allow the partial extension of dipoles during functional testing.

7.5.3 Subassembly Fixtures.

Center-Body Assembly Fixture. These fixtures, one upper and one lower, are used for assembly of the following units:

a. Docking ring assembly.

b. Solar panels, hinged.

c. Propellant tanks.

d. Jet module.

e. Tether drum mechanism.

f. Upper and lower satellite interconnecting structure.

g. Upper and lower satellite structure.

h. Fixed boom sections.

i. Boom drive motor and speed reducer assembly.

j. Electrical harnesses.

k. Electrical and electronic components.

The procedure is to assemble satellite structural members and fixed panels to fixed boom sections and interconnect structure assembly, etther tape mechanism and boom drive mechanism; assemble propellant tank and jet modules; assemble harnesses and electrical/electronic components; and install docking ring assembly.

Boom-Section Assembly Fixture. This fixture is used to assemble the following units:

a. Boom caps.

b. X-braces (boom web).

c. End members.

d. X-brace reinforcement.

e. Drive gear and tape reel.

f. Rollers.

The procedure is to position boom caps by means of roller groove. The caps are completely machined. The fixture is then adjustable by movable cap locators to accommodate all boom sizes. Boom braces are then located, and members reinforced and completely fastened. Drive gear and rollers are located and adjusted for mating the boom. This fixture will function as a gage for maintaining precise adjustment for boom components.

Fixed boom section, center body, and the section attaching to the dipole head will be built up in the boom assembly fixture.

Dipole Head Assembly Fixture. This fixture is used for the following units:

a. Dipole-head structure assembly.

b. Tank ring support bracket.

c. Drive gear assembly.

d. Dipole assembly.

e. Propellant tank ring.

f. Propellant tank quick disconnect.

The assembly fixture is used to assemble structural members of the dipole head assembly, locate dipole assembly hinge and attach pin assemblies (coordination for interchangeability), drive gear assembly, tank ring support brackets, and tank ring.

Boom sections, previously built-up, should be located and attached at this time. Dipole assemblies will not be installed here, although a check gage is desirable to ensure interface mating points are controlled. Dipole drive motor and gears are actuated for physical check.

Propellant Tank Weld Fixture. A weld fixture is used to locate fittings and ring for welding nozzle support fittings and for coordination of pin locations and ream full-size after welding.

Jet Assembly and Replacement Module. Procedure for assembly of following units is as follows:

a. Nozzle support structure.

b. Nozzle solenoid valve.

c. Quick disconnect and automatic shutoff.

d. Propellant regulator.

Assembly nozzle solenoid valves regulator, quick disconnect, and tubing to nozzle support structure. (No special tooling required.)

Connect tube from propellant quick disconnect/automatic shutoff to regulator at final assembly.

Assemble components of the replacement module in a similar manner.

Nozzle Support Structure Weld Fixture. Used to locate tubes, fittings of structure for welding, and ream interchangeable locating points full size – after welding (tooling gage).

Replacement Module Structure Weld Fixture. Used to locate tubes, fittings for welding and maintain "plug-in" point coordination (tooling gage).

Nozzle Support Structure Tooling Gage. Required to control critical interchangeable attach points for fabrication of the above weld fixtures.

<u>Metal Bond Fixture</u>. Used to bond skin and corrugation to form solar panel structure.

<u>Solar Panel Assembly Fixture</u>. Used to locate bonded panel and Z stiffeners for fastening and guide to drill holes in Z's for electronic package and component mounting.

7.5.4 <u>Manufacturing Facilities</u>. The basic facility requirements for fabrication, assembly, and checkout of the structure and various systems consist primarily of conventional sheet metal and machine shop equipment. Typical equipment that should be available include a horizontal mill with minimum 10-ft bed for machining boom cap grooves, punch press for blanking boom web sections (these parts could be routed), bonding press for 36-in. square solar panel structures, TIG or electron beam welders, and electrical cable/electronic package fabrication and test equipment.

Tool manufacturing tasks would require use of only standard machine tools. The air-bearing table for supporting the assembled boom during extension and retraction cycles could be fabricated with such equipment.

Building space requirements are of two types. Federal Standard 209 Class 100,000 or 10,000 clean-room space is needed for electronic fabrication and subassembly. Standard high-bay factory space is adequate for structural assembly and checkout of systems and subsystems, and the evaluation of capabilities available to industry. Coordination is continually maintained with the small business liaison officer so that small businesses may be given an equitable opportunity to participate in the program. Further, make-or-buy administration participates with the material department in reviewing products and capabilities presented by outside vendors to ensure recognition and knowledge of alternative industry sources.

The make-or-buy structure makes full use of products of reputable manufacturers regularly engaged in commerical production of equipment and components required for this program.

7.5.5 <u>Material Handling and Packaging</u>. Material handling and packaging is controlled by National Aerospace Standards (NAS). These standards establish the methods, materials, and devices to be used throughout the procurement, receiving, manufacturing and shipping phases of the program. Supplier packaging standards (NAS) applied to all procurement initiating documentation ensure that material is packaged for damage-free delivery to the plant, and the material is packaged so that maximum use of supplier packaging is made during receiving, receiving inspection, and storage functions.

In-plant handling and packaging standards are used as material flows from receiving, through receiving inspection, stores and manufacturing cycles. Manufactured detail parts are similarily protected during fabrication and temporary storage. A handling and shipping device is used for in-plant movement and shipment of the antenna in its packaged configuration.

Preparation for delivery instructions provide protection against damage and degradation during shipment to destination. For delivery of antenna to the destination the handling/shipping device is secured to a skidded base suitable for handling by crane or fork-lift truck. The antenna is shrouded with a barrier material to exclude dirt or other foreign contaminants. Container sides, ends, and top are provided to protect against damage. The feed support struts, explosive hardware, battery, and miscellaneous hardware are packaged and shipped in separate containers. Containers are marked in accordance with MIL-STD-129, including hazardous warnings and shipping piece numbers, to ensure proper handling and ready identification at destination.

7.5.6 <u>Make or Buy.</u> Make-or-buy policy and directives result from review of customer policies, government regulations, the technical and functional requirements of systems and subsystems, and the evaluation of capabilities available in industry. Coordination is continually maintained with the Small Business Liaison Officer so that small businesses may be given an equitable opportunity to participate in the program. Further, Make of Buy Administration participates with the Material Department in reviewing products and capabilities presented by outside vendors to ensure recognition and knowledge of alternative industry sources.

The make-or-buy structure makes full use of products of reputable manufacturers regularly engaged in commerical production of equipment and components required for this program.

7.6 <u>TEST PLAN.</u> A test program will be conducted to provide design reliability and quality assurance that the equipment will survive the pre-launch, launch, and space environments with no malfunctions or deteriorating effects; deployment can be effected with the planned amount of EVA; and that the structure and equipment will operate within specified tolerances. Testing will be divided into the following three types of tests:

a. Development tests.

b. Qualification tests.

c. Acceptance tests.

Component, subsystem, and system level testing will be performed on the following test articles:

a. Subsystem test articles.

b. Engineering model (or prototype).

c. Flight article.

Figure 7-4 identifies the proposed tests.

Figure 7-4. Test Plan Summary

TEST ARTICLES	SYSTEMS ANALYSIS	STATIC & DYN. LOADING / STRUCTURE INTEGRITY	THERMODYNAMIC EVALUATION	FUNCTIONAL TESTING ELECTRICAL/MECHANICAL	FAILURE MODE ANALYSIS	MAINTENANCE CAPABILITIES	EXAMINATION OF PRODUCT	FUNCTIONAL TEST SATISFACTORY PERFORM.	TEMPERATURE, ALTITUDE, HUMIDITY	VIBRATION PACKAGED SHIPPING & BOOSTER	ACCELERATION TEST BOOST PHASE	SHOCK TEST HANDLING & PACKAGED	PACKAGED RAIN ENVIRONMENTS	SAND & DUST	SALT ATMOSPHERE	MOISTURE AND FUNGUS RESISTANCE	THERMAL/VACUUM	SOLAR RADIATION	VIBRATION-DEPLOYED SINE/RANDOM	DEPLOYMENT TESTS ZERO G SIMULATION	RF TESTING	EMI TESTING	ACOUSTICAL NOISE TESTING	LIFE TESTS	STATIC LOADING LIMIT	PROOF TESTS
COMPONENTS																										
STAR TRACKER							X	X	X	X	X	X	X	X			X	X	X	X	X	X	X			X
HORIZON SEEKER				X			X	X	X	X		X		X	X	X	X	X	X	X	X	X	X	X		X
DIGITAL COMPUTER				X			X	X	X	X	X	X				X	X	X	X	X	X	X	X	X		X
ATTITUDE UNIT-GYROS, AUTOPILOT					X		X	X	X	X		X					X	X	X	X	X	X	X			X
RANGING AND DATA LINK, LOGIC UNIT				X			X	X	X	X			X	X	X	X	X	X		X	X	X	X	X		X
TELEMETRY RECEIVER & TRANSMITTER					X		X	X	X	X			X	X	X		X	X		X	X	X	X			X
DIPOLE MODULES				X			X	X	X	X			X	X	X		X	X		X			X			X
TLM ANTENNAS					X		X	X	X	X							X	X		X			X			X
RADIOMETER POWER DETECTORS	X			X	X	X	X	X									X	X		X	X	X				X
ACTUATION MOTORS, DIPOLE, BOOM, TETHER				X			X	X				X	X	X	X	X	X	X		X					X	X
SOLAR CELL PANELS				X			X	X	X		X	X					X	X	X	X				X		X
BATTERIES				X			X	X	X		X	X					X	X	X	X						X
PRESSURE REGULATORS			X	X			X	X	X		X	X					X		X	X						X
ACTUATION VALVES			X	X			X	X	X		X						X		X	X						X
JETS							X	X									X			X						X
SUBSYSTEMS																										
ATTITUDE CONTROL SYSTEM				X		X	X	X	X								X			X				X		X
POWER SYSTEM				X		X	X	X																X		X
GUIDANCE SYSTEM				X		X	X	X																		X
DATA LINK ELECTRONICS				X		X	X	X																		X
DATA STORAGE				X	X	X	X	X											X	X						X
DIPOLE, BOOM, TETHER ACTUATION							X	X											X							X
SYSTEMS/ASSEMBLIES																										
MAIN BODY ASSEM. SATELLITE I & II	X	X	X	X		X	X	X	X								X			X					X	X
TRIANGULAR TELESCOPING BOOM ASSEM.	X	X		X		X	X	X												X					X	X
DIPOLE HEAD ASSEM.					X	X	X	X												X						X
TETHER TAPE ASSEM.	X	X	X	X		X	X	X																	X	X
MOUNTING ADAPTER ASSEM.	X	X				X	X	X																	X	X
DOCKING ASSEM.	X	X				X	X	X																	X	X

7-14

7.6.1 Development Tests. Development tests will be conducted on the components and subsystems in the prototype stages in order to evaluate the suitability of the units for use in the flight article.

7.6.1.1 Component Tests. Some of the basic components that will be subjected to development type of tests include the following:

a. Digital computer.

b. Logic unit.

c. Dipole modules.

d. Radiometer power detectors.

e. Boom actuation mechanism.

f. Tether actuation mechanism.

g. Boom structural components.

Stress analysis, structural integrity, static, and dynamic loading tests will be performed on the structural components such as truss members and deployment mechanisms. These tests will verify the basic design limit loads and the stiffness, weight, and deflection characteristics of the specimens.

Thermodynamic evaluation tests will be performed on various electrical and mechanical components in order to demonstrate the performance under temperature, load and input power conditions. Thermodynamic evaluation will be made of such components as the radiometer power detectors, actuation motors, and attitude control jets.

Functional testing will be performed on all electrical, mechanical, and structural components in order to verify satisfactory operation of the test specimen for final design requirements.

Each component to be tested will be evaluated for maintenance capabilities while in the space environment as well as for ground conditions. Maintenance of faulty equipment will be required by EVA using the following replacement techniques:

a. Additive.

b. Replacement.

c. Regeneration.

The above replacement techniques are expected to be required on components as follows:

a. Additive:

                           Solar Cell Panels
                           Dipole Modules
                           ACS Modules: Jets
                                                 Control Valves
                                                 Regulators
                           Batteries
                           Electronic Packages

b. Replacement:

                           Complete Dipole Head Assembly
                           Dipole Actuation Motor
                           Boom Actuation Motor
                           Tether Actuation Motor

c. Regeneration:

                           Refilling of attitude control system tanks from tanks within the service module.

7.6.1.2 <u>Subsystems Tests.</u> Initial subsystem tests will be performed on prototype and engineering test models. These tests are of an evaluative nature and are also used to check out procedures to be employed in subsequent testing of the flight articles. The major subsystems to be evaluated will include the following:

a. Attitude control.

b. Power.

c. Guidance.

d. Data link electronics.

e. Data storage.

f. Dipole, boom, tether actuation.

Full-scale tests will be conducted on the above.

The subsystem assemblies will be mounted on low-frequency exciters through load transducers and sensitive low-frequency accelerometers will be attached to various components and structural members of the subsystem assembly. Vibration forces will be introduced and failure modes will be defined from the acceleration response. While employing the low-frequency, light-weight exciters, damping will be determined with the logarithmic decrement technique from response decay records obtained after the exciter armature circuit is opened. Data obtained will be integrated into calculation for the complete full-scale structure.

Functional operating capabilities will be performed on the subsystem assemblies in the following areas:

a. Electrical tests.

b. Mechanical tests.

c. Assembly tests.

d. Deployment tests.

The above tests will be performed on each subsystem as an independent unit, wherever possible.

Each electrical and mechanical subsystem will be operated and the performance monitored under ambient conditions. Where applicable, the subsystems shall be assembled in steps, demonstrating the proper action during deployment at each phase of assembly.

Subsystems will be deployed under room ambient temperature, simulating zero-g conditions by suspension from overhead members with compliant systems attached to the most massive specimen sections. Deployment force and accelerations will be measured. Overloads and out-of-tolerance conditions are applied to determine subsystem responses and recovery capability.

7.6.1.3 **System Tests.** Upon completion of the individual subsystem tests, the subsystems will be integrated into functional system/assemblies. Testing of the systems is conducted to show that the simultaneous operation of all system equipment does not produce improper operation of each subsystem or component. The major assemblies to be evaluated will include the following:

a. Main body - Satellite I & II.

b. Triangular telescoping boom assy.

c. Dipole head assy.

d. Tether tape assy.

e. Mounting adapter assy.

The objective of these tests is to demonstrate the capability of each system to withstand the launch and flight environmental conditions.

Structural vibration tests will demonstrate the structural integrity of each system. The tests will consist of vibration in three axes at design g-levels which will be in excess of those anticipated during the launch and flight phases.

Dynamic stability testing will consist of an operational test of the Attitude Control System, unrestrained, mounted on air-bearings and will demonstrate the response of the ACS to the planned program missions. The flight configuration electronic components installed for the test will be slaved to receive external stimuli simulating the flight environment during actual mission. A closed loop rf system will be used to receive signals from the test package.

The ACS will be monitored for stability, maneuverability, and orientation.

Integrated systems operational testing will be performed on each complete system with all fight equipment installed. System functional testing will exercise all electronic components, valves, switches, etc. through a typical flight mission.

Weight and balancing each system with its full complement of flight equipment is conducted to optimize the cg location and to minimize disturbing torques during launch.

7.6.2 Qualification Tests. Qualification tests will be performed or verified on each component of the complete experiment. Procured components which have been previously qualified by the vendor will be accepted only after the vendor's test has been reviewed by the cognizant engineering design group.

Qualification tests will be performed to formalized test procedures and will be witnessed by quality control inspection and verified by the customer.

Detailed test procedures covering each component to be qualified will be written by the contractor and submitted to the customer for written approval before initiation of the test program. The test procedures will include operating requirements, testing tolerances, proof cycles, step-by-step sequence of testing events, schematics of each proposed test setup, block diagrams of proposed instrumentation, and data sheets.

Qualification tests will be conducted mostly at the component level. Testing will be conducted under various environmental conditions and as a minimum will include temperature, vibration, acceleration, thermal vacuum, solar radiation, deployment tests, electrical tests, and proof tests.

7.6.2.1 Component Qualification Testing. The objective of these tests is to demonstrate each components capability to perform a required function of the complete system. Each component is functionally operated under various environmental conditions over and above the ground and flight conditions, such as increased temperatures, accelerations, shocks, and extended life tests. Component qualification testing can be divided into three groups:

a. Electrical components.

b. Mechanical components.

c. Structural components.

Qualification testing of electrical components may utilize more than one experiment component as support equipment to perform operational functions during tests. Instrumentation equipment monitoring operating parameters shall be calibrated prior to the start of testing. During use, calibration surveillance and service continues with the "valid decal" requirement.

Some of the basic electrical components that will be subjected to qualification tests are as follows:

a. Star tracker.

b. Horizon seeker.

c. Digital computer.

d. Gyros and autopilot.

e. Logic unit.

f. Telemetry receiver and transmitter.

g. Dipole modules.

h. TLM antennas.

i. Radiometer power detectors.

j. Actuation motors.

k. Batteries and solar cell panels.

Qualification of mechanical components will be performed to determine the physical characteristics of the components as well as the operating capabilities. The strength of the design and thermal gradient characteristics will be considered.

All mechanical/electrical components will require holding jigs and mounting fixtures for vibration, acceleration, and shock testing. All the fixtures will simulate the normal mounting attachments of the in flight conditions. Design, fabrication, and evaluation of all fixtures will be performed by the test lab. For components requiring deployment tests a suspension system for zero-g balanced-force environment will be built. The zero-g environment will be simulated by suspension from overhead members by compliant systems attached to the most massive specimen sections.

Some of the basic mechanical components that will be subjected to qualification tests are as follows:

a. Dipole actuation mechanism.

b. Boom actuation mechanism.

c. Tether actuation mechanism.

d. Pressures regulators.

e. Actuation valves.

f. Jets.

Qualification of structural components will be performed on specimens such as truss members and extension mechanisms. Dynamic and static limit loading shall be performed to verify the structural integrity of each component. Meteoroid impact will be considered as to the deterioration effects on each component. Life impulse tests shall be performed in order to verify reliability requirements. Failures during these types of tests may warrant design modifications prior to successful completion of qualification testing.

7.6.2.2 Subsystem and System Qualification. After completion of the individual component tests, the qualified components shall be assembled into their respective subsystem and system configurations. Subsystem and system level testing will be performed to verify satisfactory functional operation of all system equipment.

Some examples of system qualification testing follow.

The test specimen, packaged in the launch configuration, will be subjected to translational vibration test conditions simulating the predicted launch environment. This test will demonstrate the structural integrity of the packaged assembly.

The full-scale engineering model in the packaged configuration with all appropriate equipment, will be activated in the operating deployment mode. This will be done under simulated zero-g conditions. At full deployment, the truss members and joints will be effectively stress free. Full camera coverage will be used during deployment operations and functional instrumentation will be monitored.

After the full-scale engineering model has been deployed in the simulated zero-g environment, the distortion of the rf reflective surfaces will be measured.

7.6.3 Acceptance Tests. Acceptance testing will be performed on the flight article. Acceptance tests are basically functional tests performed to formal procedures approved by the customer and requiring quality control inspection witnessing and sell-off to the customer's in-plant representative.

Acceptance test environments will never exceed the operating design levels encountered during ground and launch conditions. Acceptance tests to be performed on the flight article are selected from the sequence of tests conducted during qualification testing.

7.6.4 <u>Test Facilities</u>. The facilities described in the following paragraphs represent only the major items that would be used in direct support of the program. Omitted are secondary support facilities such as laboratory and bench type test equipment which the typical aerospace contractor would normally have available.

Facility requirements are predicated on a production of three antennas of the crossed-H interferometer configuration. Manufacturing fixtures and equipment will serve a dual purpose in this program in that they will, in many cases, be used also during the testing phases of the fabrication and development program.

7.6.4.1 <u>Component Test Facilities</u>. Testing at the component level requires environmental chambers, vibration equipment and a vacuum chamber with a cryogenic shroud and solar radiation capability. Shielded enclosures will also be used for conducting electromagnetic interference tests.

7.6.4.2 <u>Subsystem Test Facilities</u>. Subsystem test programs will utilize all of the equipment previously identified for component testing and such additional equipment as structural loading facilities and a large vacuum chamber. The latter must be large enough to accommodate a representative segment of the antenna structure that will be activated from a packaged configuration to deployment. This operation will be conducted while the chamber is providing a simulated deep space environment, complete with cryogenic shroud and solar radiation, and the antenna is deployed by means of a zero g fixture. In addition, the selected contractor should have at his facility, or have access to, a large swimming pool. This pool will be used by potential astronauts in conducting EVA studies on the crossed-H interferometer structure while experiencing a zero g or neutral buoyant conditions. These typical activities will include assembly and repair of antenna structural members, and routine adjustments and inspections.

7.6.4.3 <u>System Test Facilities</u>. During the system checkout and evaluation program, whenever practical, manufacturing fixtures will serve a double purpose in that they will also serve as a test fixture.

Testing programs will be conducted in the environmental controlled assembly area in order to maintain desired tolerance requirements. Component and subsystem level test equipment will meet the needs of the systems test with the exception of the final rf test program.

7.7 <u>SUPPORT PLAN</u>. The crossed-H interferometer support plan summarizes the general requirements for all activities performed on the spacecraft subsequent to sell-off of the flight unit at the contractors facility. It also defines the necessary support for these activities. Typically, included are:

a. Personnel training.

b. Prelaunch activities.

c.   Range documentation.

d.   Launch operations.

e.   Orbital operations.

7.7.1   Personnel Training.   Special training is required to develop the necessary skills in mission-oriented tasks.   In addition to the detailed training program for the astronauts, some training is required for the prelaunch operations crew, the launch crew, and the orbital operations (communications and data processing) crew.

It is assumed that two three-man crews will be trained concurrently for the crossed-H interferometer mission.   Astronauts chosen for this mission will have had training in rendezvous, docking, EVA procedures, and zero-g simulation experience in underwater facilities.   Electronics or mechanical engineering background, with specific experience in radio-astronomy is desirable but not necessary.   The training program will include:

a.   rf receiving equipment theory.

b.   System familiarization.

c.   Radio-astronomy theory.

d.   EVA training exercises.

e.   EVA simulations.

f.   Malfunction detection analysis and repair.

g.   Safety procedures.

The first phase will be an intensive refresher course in long-wave antennas and receiving equipment theory.   Overlapping this is the indoctrination in use of the equipment and hardware associated with the antenna system.   The study of malfunction detection, analysis, and repair techniques will be most effective if protracted over a period of four months, spanning the time in which the various phases of the mission are simulated.   The safety procedures portion of the program will concentrate on the EVA aspects, analyzing the hazards associated with inspection, adjustment, and repair.   Emergency conditions will be simulated in an underwater facility.

A brief technical orientation and training session for range orientation is also required. This session includes presentations by the experimenter, the contractor and personnel from the orbital range.   The principal investigator or his representative describes the experiment, its objectives, and the data gathering requirements.   A range representative presents ground-station capabilities, range operating techniques and planned level of range support.   The contractor discusses the spacecraft design capabilities, command control requirements, and the planned orbital operations philosophy.

7.7.2 __Prelaunch Activities.__

7.7.2.1 __Handling and Shipping Operations.__ The crossed-H interferometer is shipped completely assembled, with experiment equipment installed, but, in the stowed condition, from contractor's facility to the Marshall Space Flight Center by government air freight. Special handling is designated. Subsequent to MSFC operations the spacecraft is again air-lifted to the launch site (KSC). Other transportation is by specially designed flat-bed trailer assembly.

The spacecraft is transported at all times in a specially designed shipping container. Solar cell arrays, batteries and operational test equipment are shipped separately and handled as delicate instruments. Installation hardware is kitted in suitable containers and secured in the appropriate shipping containers.

7.7.2.2 __MSFC Operations.__ The spacecraft and ground support equipment is shipped to MSFC for MSFN network compatibility tests and Saturn V fit checks.

a. MSFN compatibility test. The spacecraft is removed from the shipping container, visually inspected and subjected to a standardized checkout of electrical, command, telemetry and experiment subsystems. Test equipment, procedures and operations are identical to those used at the launch site. MSFC establishes the requirements for the compatibility tests and directs their performance.

b. Saturn V fit-check. The crossed-H interferometer spacecraft system, LEM adapter system and the CSM are mated to determine their mechanical/structural compatibility.

    Static weight and balance measurements and cg trim are accomplished at this time.

Following all tests, the spacecraft is re-installed in the shipping container and transported to KSC for launch operations.

7.7.3 __Range Documentation.__ The basic documentation requirements of the MSFN stations are listed below.

a. Support and Instrumentation Requirements Document. This document specifies requirements for facilities, data processing logistics, telemetry, and instrumentation support. It also furnishes a brief description of the spacecraft and provides detailed information on the characteristics of the telemetry and command systems.

b. Operations Plans. The operations plan is prepared by NASA to specify requirements for injection and early orbital support. The contractor inputs to this document include the launch operations test plan, the spacecraft operations notebook and nominal command schedule.

The launch operations test plan pertains primarily to the launch range operations and requirements. It specified procedural and equipment interfaces and provides an operations schedule from the time of shipment of the launch site to orbital injection. However, it also furnishes information pertinent to orbital operations such as vehicle frequency utilization, telemetry formats, predicted launch mark events and processing requirements for orbital data.

The spacecraft operations notebook includes subsystem descriptions, command control operating instructions, an orbital operations philosophy, and housekeeping data analysis. The command control instructions present a detailed analysis of spacecraft response to each of the possible commands under normal and abnormal conditions. It also defines and explains all cautions to be observed during operation. The orbital operations philosophy is a detailed discussion of the theory and reasoning used as a basis for the command schedule. The purpose is to provide the MSFN spacecraft controllers with guidelines so they may utilize the maximum capability of the spacecraft without endangering its operational life. This allows flexibility and rapid response to experimenter requests. Definition and analysis of all housekeeping data and their predicted limits are included in the operations notebook. The housekeeping data crtical to spacecraft and experiment health are designated as red-line or go/no-go functions and their permissible limits defined.

The spacecraft nominal command schedule provides predicted look angles, acquisition times, and sequence of commands to be transmitted for each ground station.

c. Spacecraft Telemetry Test Tape. A voice annotated magnetic tape of the telemetry composite video signal recorded during simulated normal and abnormal orbital acquisitions will be provided to the range for operational training and data processing tests. A script of the tape contents noting time and duration of commands will also be furnished.

d. Range Compatibility. Spacecraft-to-ground station compatibility may be verified by arrangements with the MSFN Satellite Operations Center for the applicable NASA stations to acquire and operate the spacecraft in orbit. The telemetry test tape could also be used for compatibility checks.

e. Test Evaluation Report. A test flight report will be prepared for the crossed-H interferometer spacecraft system. This report includes evaluation of the propulsion module through the period of orbital injection. This portion of the report consists of a quick-look diagnostic evaluation of MSFN - and KSC supplied data. Evaluation of the spacecraft performance throughout the mission will be based on MSFN-supplied reduced data. It includes sufficient detail to evaluate overall performance against predicted criteria for each of the subsystems.

7.7.4 Launch Site Operations. At the launch site the spacecraft undergoes inspection, checkout, Saturn V assembly and launch operations.

7.7.4.1 Sequence of Operations. A summary of the sequence of events is given in Figure 7-5. Shipping and receiving of all crossed-H interferometer equipment are accomplished per published shipping instructions listing all deliverable items with inspection and storage instructions for each. These functions are accomplished by the organization and facilities established to support previous AAP operations.

A clean, environmentally-controlled area is provided for storage, assembly, and test of the satellite and experiment equipment.

Engineering confidence tests are performed on all spacecraft subsystems to verify proper operation after shipment and prior to experiment interface checks. These tests are accomplished by cycling the spacecraft through its operating modes while recording and verifying data per the spacecraft functional checklist. If an anomaly is noted, the particular subsystem is tested per the appropriate section of the crossed-H spacecraft subsystems acceptance test.

In addition to these confidence checks, two formal engineering procedures are performed: 1) solar array electrical test, and 2) electrical storage battery charge procedure.

Engineering confidence tests verify the mechanical and electrical interfaces between the spacecraft support subsystems and the experiment payload. Interface tests are performed with both the flight and the backup subsystems.

Electrical interfaces are checked in all operating modes. These tests verify proper response of the spacecraft and experiment payload to external commands and to internal logic signals. Data from these tests are reviewed for quality and compared to data from the spacecraft acceptance test.

Upon completion of interface checks the spacecraft is assembled to prepare for mating with its payload adapter assembly. The assembly procedure includes:

a.   Thorough cleaning of the spacecraft/adapter interface.

b.   Installing the flight batteries.

c.   Verifying proper mating of each spacecraft electrical connector.

d.   Installing solar panels.

A formal engineering systems test is performed after preflight assembly and prior to mating with the payload adapter assembly. This test duplicates, as closely as the fully assembled configuration will permit, the spacecraft systems acceptance test performed at the factory. The data are analyzed to verify proper operation of all subsystems and compared with like data recorded during launch site confidence tests and the factory systems acceptance test.
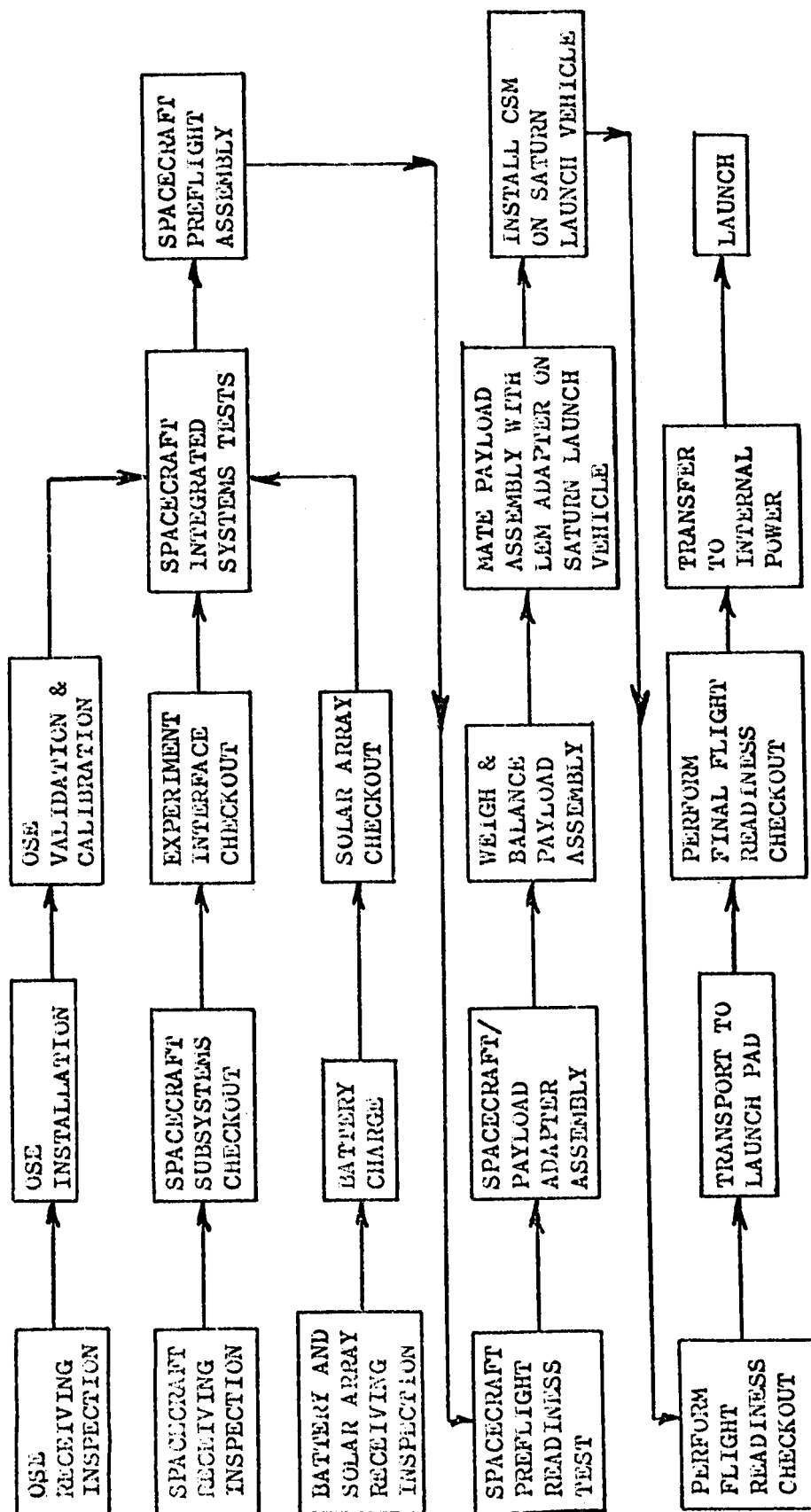
Figure 7-5. Launch Site Operations Flow Chart

After completing the preflight readiness test the weight and balance of the assembled payload is checked to verify launch configuration. The payload is then installed in the LEM adapter and mounted atop the Saturn launch vehicle.

The flight readiness checkout procedure includes four sections: 1) spacecraft system checks, 2) experiment system checks, 3) installation, checkout, and interface verification, and 4) final spacecraft flight preparation. The subsystems checks again verifies proper response of the spacecraft and experiment payload command control logic to external commands and internal logic signals as well as providing data for qualitative analysis of the telemetry and electrical power systems.

The total vehicle is then transported from the VAB, by means of the crawler-transporter, to its final position on the launch pad. There, final flight readiness checkouts are performed in conjunction with the launch vehicle countdown. Shortly before launch the crossed-H spacecraft is transferred to internal power to sustain standby loads until final is achieved.

7.7.4.2 _Launch Site Procedures._ All operations at the launch site are performed under the surveillance of MSFC quality control personnel to ensure maintenance of system configuration and performance integrity. An engineering log of all launch site activities is maintained by the systems engineer. Operations are divided into two categories: 1) formal engineering tests, and 2) engineering confidence tests. The formal engineering tests are conducted by cognizant design engineers in accordance with published procedures specifying step-by-step sequential operations and witnessed by a quality control inspector. The engineering confidence tests are also conducted by cognizant design engineers in accordance with functional checklists specifying required data and monitored by a quality control inspector.

Table 7-1 lists typical launch site procedures applicable to the crossed-H spacecraft and the purpose of each.

7.7.4.3 _Launch Site Facilities._ (Reference Douglas Report SM-47274, Saturn V Payload Planner's Guide.) Launch operations for the Saturn V vehicle are conducted at Complex 39 and utilize the mobile, or off-pad-assembly, concept. This concept, which provides for a greater flexibility and launch rate than on-pad assembly, employs four basic operations: 1) vertical assembly and checkout of the Saturn V on a mobile launcher in a controlled environment, 2) transfer of the assembled and checked-out-vehicle to the launch pad on a mobile launcher, 3) automatic checkout at the launch pad, and 4) launch operations by remote control from a distant launch control center. The major units involved in this concept are the Vertical Assembly Building (VAB), Launch Control Center (LCC), Mobile Launcher (ML), Mobile Service Structure, Crawler-Transporter, Launch Pads, and High Pressure Gas Facility. Figure 7-6 shows a schematic illustration of the complex.

Table 7-1. Spacecraft Launch Site Procedures

TITLE	PURPOSE
Spacecraft Shipping Instructions	Provide listing of all deliverable items with instructions for shipping and storage.
Test Equipment Validation	Provide instructions for system qualification of special test equipment. (Standard test equipment will be calibrated by local calibration or precision equipment laboratory.)
Spacecraft Subsystems Check list	Engineering confidence test for support systems.
Solar Array Electrical Test	Test of open circuit voltage and short circuit current of each series string with a fixed illumination source. Also checks forward and reverse impedance of isolation diodes.
Battery Charge	Provides instructions for formation charge, standard full charge or partial charge of the spacecraft electrical storage battery.
Spacecraft Final Assembly	Prepares spacecraft for mating with final stage.
Preflight Readiness	Final spacecraft systems test.
Flight Readiness	Performs functional subsystem tests on launch pad and provides instructions for final flight preparations.
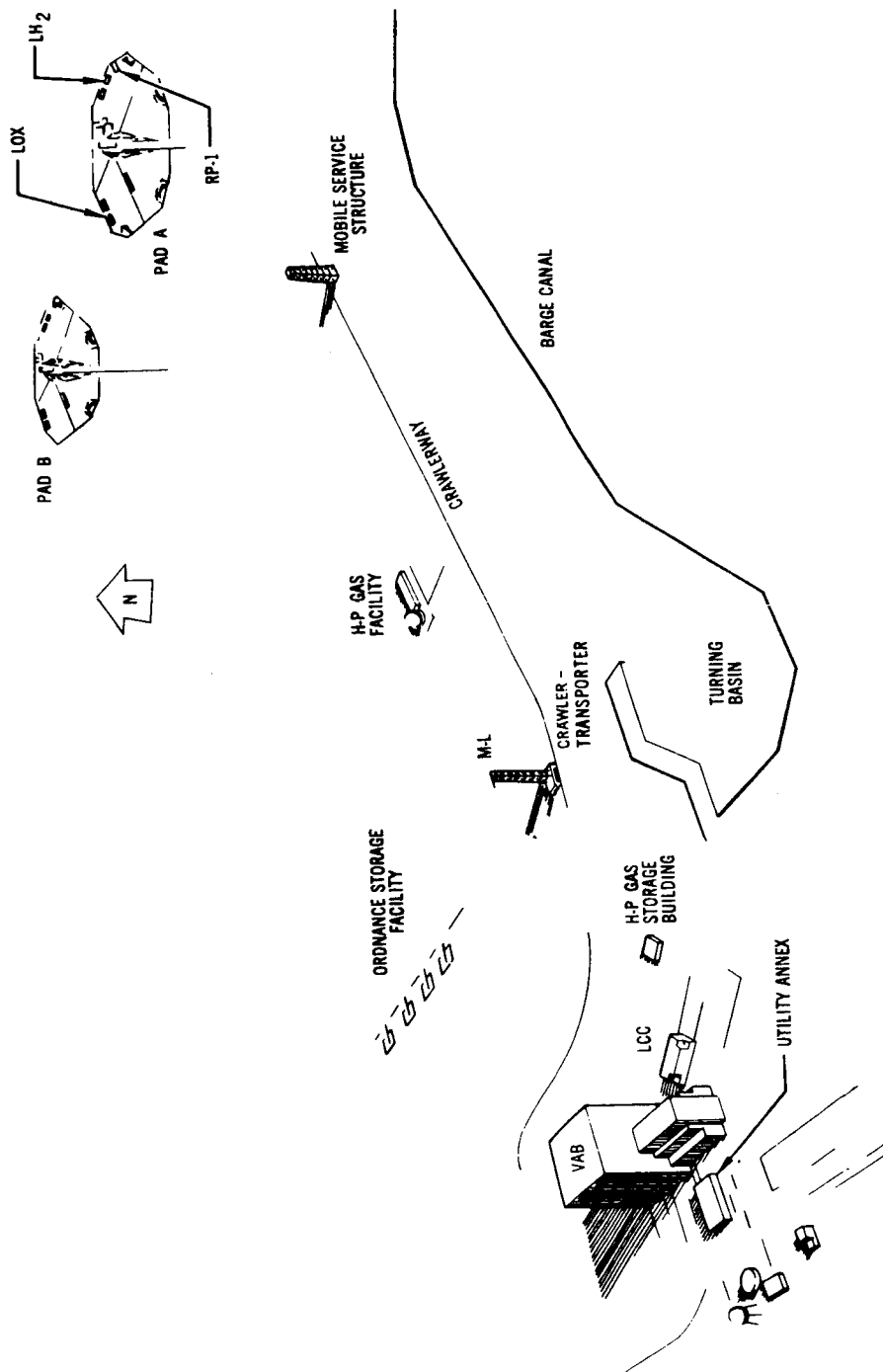
Figure 7-6. Saturn V Launch Complex

7.7.5 <u>Mission Operations.</u> Flight mission operations are controlled by NASA.

<u>Boost and Orbit Injection.</u> Definition of the flight sequence from liftoff through injection of the CSM/crossed-H spacecraft payload into synchronous orbit will result from trajectory shaping and optimization for the specific payload involved.

<u>Orbit Operations.</u> Initial orbital activities utilize the capabilities of the astronaut crew to the fullest extent possible. Flow charts for astronaut participation and orbital operations are included in Appendix A.

7.8 <u>SCHEDULE.</u> The schedule, Figure 7-7, indicates a five-year development program, following the current Phase A/B study, and resulting a mid-1972 launch of the crossed-H interferometer. Some overlap in prototype assembly and test with the equivalent tasks of the operational system assembly and test is indicated. In support of a later launch date, an evaluation and design modification cycle can be incorporated following completion of the prototype tests, and preceding initiation of flight article assembly.

7.9 <u>COST ANALYSIS</u>

7.9.1 <u>Introduction and Ground Rules.</u> Total system costs were developed for each of the three experiments selected in the earlier phases of the study and that were considered in detail in Task 4. The funding requirements for the crossed-H interferometer antenna are presented herein. Included are nonrecurring cost, recurring cost, vehicle support, and facility costs.

The cost estimating task for this experiment system was carried out in accordance with guidelines provided by MSFC as outlined below:

a. Cost estimates are to be developed for the experiments in accordance and compatible with, the general format outlined in "Cost Estimation for Future Programs" (ASO 12 May 1967).

b. The level of detail of the costs will be dictated by the level of experiment definition attained and the time allocated to this portion of the task.

c. Cost estimates are to be made for the space structure and support hardware and experiment system integration only. Costs will not be included for the launch vehicle, Apollo CSM or other spacecraft, mission support, mission operations, AAP payload integration, or subsequent flights for rendezvous and/or experiment refurbishment.

d. The program to be costed will include one flight experiment vehicle only, with no backup flight article.

Figure 7-7. Schedule

1. CONTRACTOR'S STUDY EFFORT
2. NASA REVIEW AND ANALYSIS
4. RFP PREPARATION
5. PROPSAL PREPARATION & ANALYSIS
6. CONTRACT AWARD
7. CONTRACTOR'S SYS. DEFINITION EFFORT
8. GUIDANCE & ATTITUDE CONTROL SUBSYSTEM
9. PRELIMINARY DESIGN & SPECIFICATIONS
10. MANUFACTURING & TEST REQUIREMENTS
11. RADIOMETRIC SUBSYSTEMS
12. PRELIMINARY DESIGN & SPECIFICATIONS
13. MANUFACTURING & TEST REQUIREMENTS
14. NASA REVIEW & ANALYSIS
15. NASA APPROVAL & SELECTION
17. PHASE C DESIGN
18. NASA DRAFT PHASE "C" PLAN
19. PREPARE RFP
20. PROPOSAL PREPARATION & ANALYSIS
21. CONTRACT SELECTION & CONTRACT AWARD
23. SYS.-DETAIL DESIGN DEVELOP.& INTR.
    GUIDANCE & ATTITUDE CONTROL
24. RADIOMETRIC
25. STRUCTURAL/MECHANICAL
26. FACILITIES - DESIGN/CONSTRUCTION
27. COMPLETE SYSTEMS DESIGN DATA
28. NASA REVIEW & ANALYSIS
29. NASA APPROVAL & SELECTION
32. PHASE D DEVELOPMENT & OPERATION
33. NASA DRAFT PHASE "D" PLAN
35. CONTRACT NEGOTIATION & AWARD
36. DESIGN OPERATIONAL SYSTEM
37. FAB. DETAIL PARTS/COMPONENTS
38. PROTOTYPE-ASSEMBLY & TESTS
37. OPERATIONAL SYSTEM-ASSEMBLY & C/O
38. - TEST & CERTIFICATION
39. - SHIP TO KSC
40. ASTRONAUT TRAINING
41. LAUNCH OPERATIONS & SERVICES
42. FLIGHT OPERATIONS & SERVICES
43. DATA ACQUISITION, REDUCTION & ANALYSIS
44. SCIENTIFIC EXPERIMENTS
45. SPACE VEHICLE REFURBISHMENT

e. The most detailed costs will be developed for the Parabolic Expandable Truss Antenna Experiment. The next most detailed costs will be generated for the Crossed-H Interferometer Long Wave Radio Astronomy Antenna. The Focusing X-Ray Telescope Experiment is expected to have less cost detail than the first two experiments.

f. In addition to total system costs, funding requirements will be presented in accordance with NASA phased project planning guidelines. These fiscal year cost estimates will include Phase B (Definition), Phase C (Design), and Phase D (Development).

g. In addition to total dollars, the system cost estimate is to include a breakout of labor in man years and material costs.

Further ground rules that were used in this analysis include the following:

a. 1967 dollars were used throughout both for labor and material.

b. Present manufacturing and test facilities are assumed adequate and available for the conduct of this program with the exception of these unique or new facilities included in the cost estimate.

c. Fully developed and flight qualified hardware components were utilized wherever possible.

d. The program is assumed to be a normally paced (non-crash) development program. Labor costs are based on a single shift operation.

e. Costs were based on the flight date of 1972 shown in the schedule presented in Section 7.8.

f. NASA in-house costs are excluded.

7.9.2 Cost Estimating Procedure. In general, costs were estimated for 1) nonrecurring or development phase of the effort, 2) recurring cost or flight hardware unit cost, 3) ground support equipment (GSE and STE), and 4) facilities.

Nonrecurring Cost. The nonrecurring cost includes all research, development, design, analysis and test including all development hardware necessary to fabricate and fly the operational experiment.

The nonrecurring cost as defined herein includes only those costs up to the point where fabrication of the flight article is initiated. The nonrecurring cost is added to the recurring (unit) cost to obtain total program cost. This procedure was used to give an incremental unit cost for the flight unit in a one-vehicle program.

The system and subsystem definition and the development plans were reviewed and analyzed to determine general task requirements at the major subsystem level.

Manpower requirements were then estimated for these tasks and for overall system integration task.

The development plan and the test plan provided the basis for defining the development and test units, other test hardware, and associated tooling. Costs were then developed for both material and labor.

Composite labor rates were applied to manpower requirements and summed with the materials cost and overhead rates for the total nonrecurring cost estimate.

Recurring Cost. The recurring cost includes the incremental unit cost of the experiment, test operations and checkout, and spares.

The design definition at the system and subsystem level was reviewed and analyzed and a major component list prepared. Costs were then estimated for purchased items. In the case of manufactured items, material and labor costs for fabrication and assembly including the appropriate tooling, quality control, etc. were estimated. System/ subsystem test and checkout labor was estimated based on the complexity of the experiment and appropriate costs developed from available cost analogs. Appropriate factory overhead, material burden, and G&A overhead factors were then added to provide total incremental unit cost.

Spares are estimated as a percentage factor of total estimated unit cost because of lack of definition in this area.

Vehicle Support. Cost estimates were made both for GSE and Special Test Equipment (STE) at a relatively high level of aggregation because of the lack of design definition of the equipment in this category.

Facilities. The design definition and the manufacturing and test plan for each experiment was examined to determine the requirements for new or unique and unusual facilities in each of the areas of manufacturing, test, and operations, and served as the basis of the cost estimates for these required facilities.

7.9.3 Cost Uncertainties. The confidence limits of the cost estimates presented in this report are believed to be compatible with the level of definition of the subsystems, components, and with the development plan available at the time of the cost estimating effort. These estimates are to be regarded as area estimates and are based on varying degrees of definition. In some areas, such as some of the experiment instrumentation, only cost allowances were made. These areas are discussed and identified below. Cost estimates in more detail as well as greater confidence will be developed in Phase B (Definition) of the phase project planning cycle.

7.9.4 **Program Cost.** Summary costs for the crossed-H interferometer antenna are presented in Table 7-2 and detailed breakouts in each of the areas of nonrecurring, recurring, vehicle support, and facilities are presented in Tables 7-3 through 7-5. Funding requirements by fiscal year for Phase B, C, and D are presented in Table 7-6.

The total nonrecurring cost for this experiment is presently estimated to require $19.8 M for the program design and development. A recurring cost of $7.3 M, is required for the fabrication of a single flight article. The total program cost is estimated at $29.0 M, including the vehicle support, and facilities, and Phase B (at $1 M). The assumptions and uncertainties are discussed below.

Hardware costs were developed on the basis of purchased components and materials and on manufactured items. Labor estimates were made for the overall system tasks, and the design, analysis, test, integration, and fabrication processes for the "make" items - primarily the structure. The costs presented herein were based on one complete flight article, a prototype consisting of a complete lower satellite plus an upper satellite center body only (no booms), and on component hardware used in subsystem and component testing. Some of the subsystems on the prototype equipment will not be flight configurations. For example, because of the high cost of the solar cells and their installation, many dummy panels will be utilized in the prototype.

The components selected for costing purposes are current off-the-shelf, space qualified units whenever possible. These components were selected for performance that either met or exceeded the required capabilities. In cases where no existing qualified components were available, costs were based on equipment with excess capability. This was done to avoid incurring a large development or qualification cost at the component level which would completely override any recurring hardware savings which might be associated with a lower performance requirement. In the case of the star trackers, costs based on trackers with 15 arc sec capability because no space qualified trackers in the one to two arc minute range are available. Similarly, the tracking telemetry, command system, and the data management/data storage system components were estimated on the basis of the similar systems in the Apollo CSM telecommunication system even though certain elements of this equipment may have excess capability. In some cases, a lower cost estimate was used to allow for the fact that only portions of the electronic system "black boxes" were required.

Certain of the component packages are not expected to be off-the-shelf in the time period of interest. The principal equipment requiring development will be the experiment instrumentation systems. This electronic system includes receivers, filters, power detectors, correlation detectors, and phasing and combining networks, and since detailed definition is not yet available, only a ball-park area allowance was made for both the unit cost and the development cost of this equipment. This equipment appears to be conventional components which involve straightforward development programs. The principal development effort for these components is for packaging and the qualification test program.

Table 7-2. Cost Summary

	Cost (millions of dollars)	
Nonrecurring Cost (Phase C & D)		20.705
Design and Development	19.785	
GSE/STE	0.820	
Facilities	0.095	
Recurring Cost (Unit Cost)		7.295
Total Program Cost		28.000

The structure appears to be relatively straightforward from the cost viewpoint, and there appears to be only an air-bearing attitude control test facility requirement for its fabrication and test. Some form of air-bearing overhead support for boom deployment tests will be required and is included herein under special test equipment. It is doubtful that a complete deployment test for each of the satellites will be possible on the ground and that these tests will have to be conducted one boom at a time and include only partial dipole deployment. If a full satellite deployment test is required, considerably more extensive STE will be required as well as additional facilities.

The cost estimates presented are believed to be representative for this program in view of the present state of definition. Upon more detailed analysis some of the cost factors may prove sensitive to further definition and to design changes. Within this context the cost may be expected to be in the tolerance range of -10% to +30% of this estimate.

Table 7-3. Nonrecurring Cost

	Engineering Design, Development and Analysis	Development And Test Hardware				Total
		Tooling	Labor	Material	Test	
Structure	0.200 (6.5)	0.090 (3.5)	0.195 (8.8)	0.200	0.150 (5.5)	0.835
Electrical Power System	0.180 (5.8)	0.010 (0.4)	0.035 (1.5)	0.965	0.175 (6.8)	1.365
Stability and Control	1.025 (32.8)	0.010 (0.4)	0.045 (2.0)	2.010	0.250 (10.0)	3.340
Telecommunications/Data	0.545 (17.2)	0.010 (0.4)	0.080 (3.5)	3.945	0.270 (10.0)	4.850
Experiment Instrumentation	0.400 (12.8)	0.010 (0.4)		4.200	0.230 (9.0)	4.840
Misc. Systems	0.030 (1.0)	0.005 (0.2)	0.060 (2.8)	0.005	0.045 (1.5)	0.145
Vehicle System (Assembly and Integration)	0.250 (8.0)	0.030 (1.2)	0.105 (6.2)		1.085 (37.0)	1.470
Tech Data/Manuals						0.440 (8.4)
System Engineering/Management						1.755 (50.8)
Training						0.715 (23.8)
Travel						0.035
TOTAL	2.630	0.165	0.520	11.325	2.205	19.790

Cost (millions of dollars)

(Labor requirements, in man years, are shown in parenthesis)

Table 7-4. Recurring Cost

	Cost (millions of dollars)				
	Fabrication And Assembly		Test And Checkout	Spares	Total
	Labor	Material			
Structure	0.205 (9.4)	0.145			0.350
Electrical Power System	0.043 (2.2)	1.165			1.208
Stability and Control	0.066 (3.0)	1.435			1.501
Telecommunication/Data	0.123 (5.8)	1.065			1.188
Experiment Instrumentation	0.088 (3.9)	0.560			0.648
Misc. Systems	0.015 (0.6)	0.005			0.020
Vehicle System (Assembly and Integration)	0.070 (3.1)		0.140 (6.2)	0.635	0.845
Sustaining Engineering					0.655 (19.9)
System Management/System Integration/Training					0.460 (15.8)
Launch Support					0.385 (12.8)
Travel					0.035
TOTAL					7.295

Table 7-5. Vehicle Support and Facilities

	Cost (millions of dollars)
Vehicle Support	
AGE	
Launch Site Support	$0.170
In-Plant Support	0.505
Total	0.675
STE	0.145
Facilities	
Manufacturing	--
Test	0.095
Operational	--
Total	$0.095

Table 7-6. Funding Requirements - Phased Project Planning

	FY1	FY2	FY3	FY4	FY5	FY6
Phase A						
Phase B	1.000*					1.000
Phase C		7.415	6.160			13.575
Phase D			8.555	5.740	0.130	13.975
Total	1.000*	7.415	14.715	5.740	0.130	29.000

*Assumes two contractors @ $500,000

SECTION 8

REFERENCES

1. Dr. N. Frank Six, Jr., Low Frequency Radio Astronomy Experiments in Space, Research Laboratories Brown Engineering Company, Inc., Technical Note R-180 Huntsville, Alabama, February 1966.

2. Space Research Directions for the Future, Part Two, Space Science Board, National Academy of Sciences, National Research Council, January 1966.

3. Significant Achievements in Space Astronomy 1958-1964, National Aeronautics and Space Administration, NASA SP-91, 1966.

4. Fred T. Haddock, Phase I Final Report Engineering Feasibility Study of a Kilometer Wave Orbiting Telescope, University of Michigan Radio Astronomy Observatory, NGR 23-005-131, October 1966.

5. J. R. Hubbard and W. C. Erickson, "A Stabilized Cross-Correlation Radiometer", IEEE Trans. on Antennas and Propagation, Vol. AP-15, No. 2, pp. 291-294, March 1967.

6. H. Ashley, "Observations on the Dynamic Behavior of Large Flexible Bodies in Orbit", AIAA Journal, 5, 3, 460, March 1967.

7. R. H. Brown and A. C. B. Lovell, The Exploration of Space by Radio, pp. 42-54, Chapman and Hall, Ltd., London, 1957.

8. G. Howell, Mathematical Model and Digital Computer Program for the Solution of Temperature Distribution Around Screen Type Tubular Gravity Gradient Elements, Convair division of General Dynamics Report GDC-DDB66-005, March 1966.

APPENDIX I

CREW FUNCTIONAL ANALYSIS
AND FAILURE PROCEDURES

The top level functions of the Crossed-H Interferometer are shown in Figure I-1. Second and third level function flow diagrams were derived for the extraction, checkout, deployment, and refurbishment. See Figures I-2 and I-3.

The "operation" function is omitted here as it is described in Figures 3-3 and 3-4. The "maintenance" function is also omitted as it consists essentially of selecting the appropriate item under "refurbish".

The function "refurbish" is treated in detail for most of the unique components of this experiment. Table I-1 shows the detail functions, crew participation, EVA, and failure considerations.
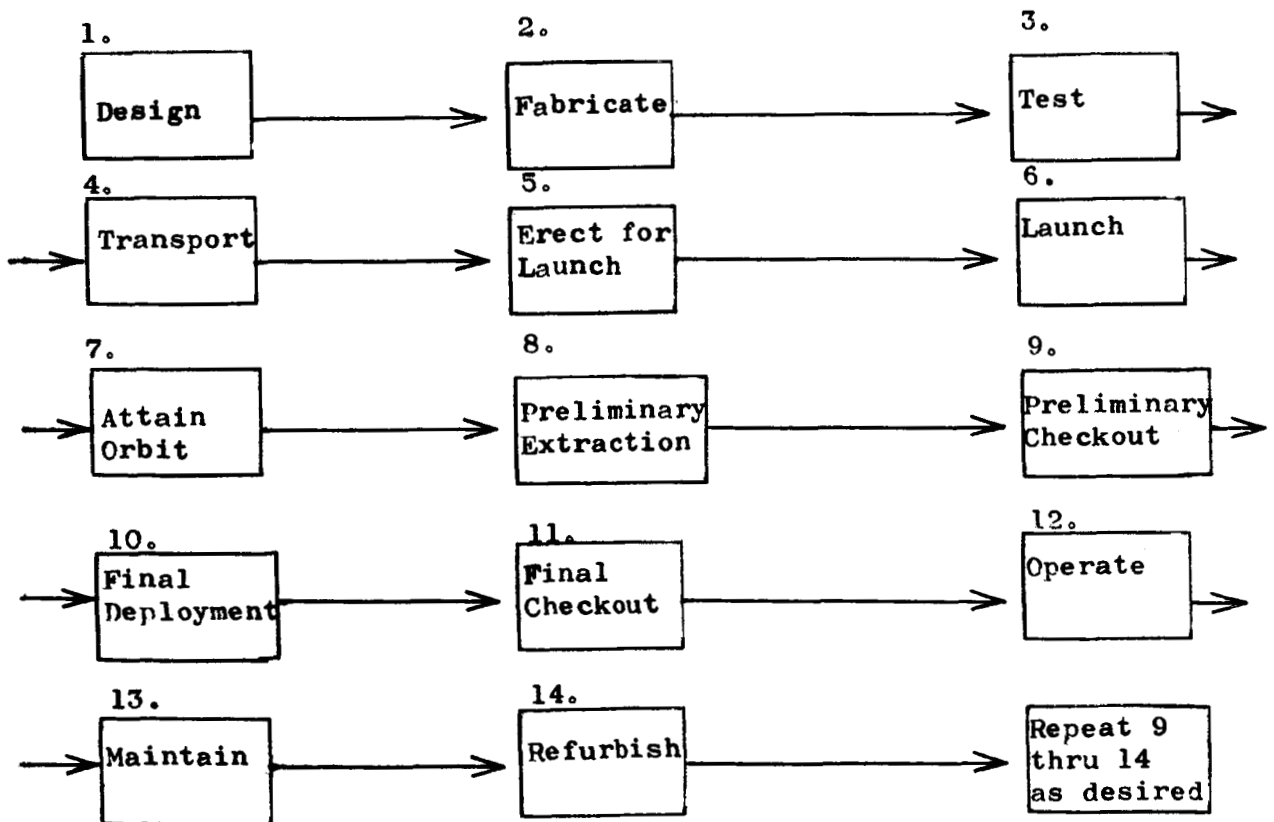
Figure I-1. First Level Functions

8. **PRELIMINARY EXTRACTION**

8.1	8.2	8.3	8.4	8.5
Release CSM	Extend SLA	Dock CSM to Experiment Assembly	Withdraw experiment assembly from SLA	Stabilize with CSM for preliminary checkout

9. **PRELIMINARY CHECKOUT**

9.1	9.2	9.3	9.4	9.5
Command TM Operation	Check Battery Output	Check TM Operation	Checkout Power Subsystem	Checkout (Preliminary) ACS

9.6	9.7
Checkout (Preliminary) RF Subsystem	Checkout Data Storage S.S.

10. **FINAL DEPLOYMENT**

10.1	10.2	10.3	10.4	10.5
Orient Antenna by Means of CSM ACS	Release Antenna Assy. from Experiment Assembly	Dispose of Experiment Assembly	Survey Antenna for Launch & Deployment Damage	Proceed to No. 11 Final Checkout

11. **Final Checkout**

11.1	11.2	11.3	11.4	11.5
Checkout (Preliminary) Tether Drive Subsystem	Checkout (Final) ACS Subsystem	Checkout Boom Drive Subsystem	Checkout Dipole Subsystem	Checkout (125M) RF Subsystem

11.6	11.7	11
Checkout (Final) Tether Subsystem	Checkout (Final) RF Subsystem	Proceed to Operate

Figure I-2. Second Level Functions (Sheet 1 of 2)

14. REFURBISH

```
14.1                           14.2.
┌──────────────────┐           ┌──────────────────┐
│  Prepare         │           │  Replace         │
│  for             │──────▷    │  Dipole          │──────▷
│  Refurbishment   │           │  Modules         │
└──────────────────┘           └──────────────────┘


14.3                           14.4
   ┌──────────────────┐           ┌──────────────────┐
   │ Replace Sole-    │           │                  │
──▷│ noid/Nozzel      │           │  Replace         │
   │ Assy.(including  │──────▷    │  Propellant      │──────▷
   │ Regulator)       │           │  Tank            │
   └──────────────────┘           └──────────────────┘


14.5                           14.6
   ┌──────────────────┐           ┌──────────────────┐
   │ Replace Elec-    │           │ Replace Electro- │
──▷│ tric Motors (2)  │           │ nic Module (4)   │
   │ & Solenoid for   │──────▷    │ at Dipole Head   │──────▷
   │ Dipole Drive     │           │                  │
   └──────────────────┘           └──────────────────┘


   ┌──────────────────┐
──▷│  Etcetera        │
   └──────────────────┘
```
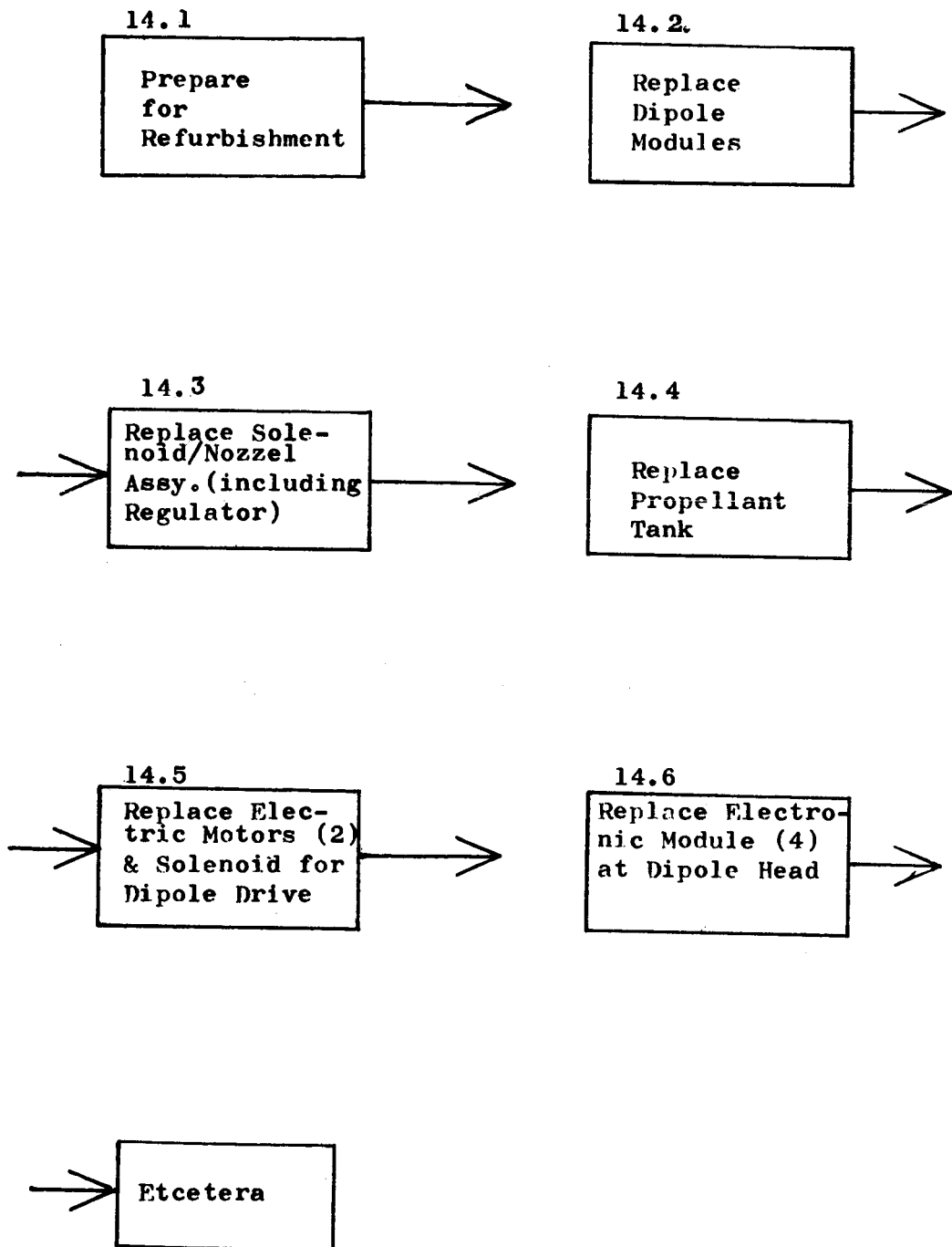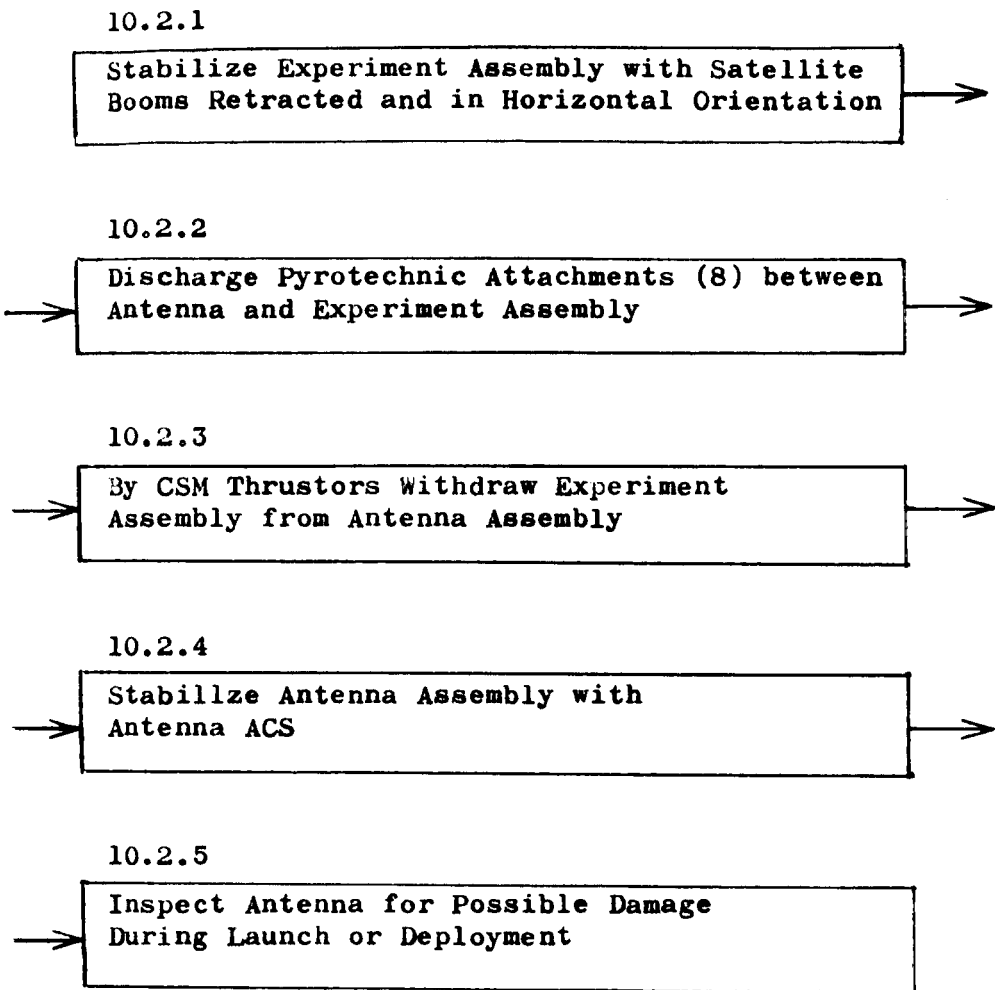
Figure I-2.   Second Level Functions (Sheet 2 of 2)

10.2 RELEASE ANTENNA ASSEMBLY FROM THE EXPERIMENT ASSEMBLY

10.2.1

> Stabilize Experiment Assembly with Satellite
> Booms Retracted and in Horizontal Orientation ➝

10.2.2

➝ Discharge Pyrotechnic Attachments (8) between
Antenna and Experiment Assembly ➝

10.2.3

➝ By CSM Thrustors Withdraw Experiment
Assembly from Antenna Assembly ➝

10.2.4

➝ Stabilize Antenna Assembly with
Antenna ACS ➝

10.2.5
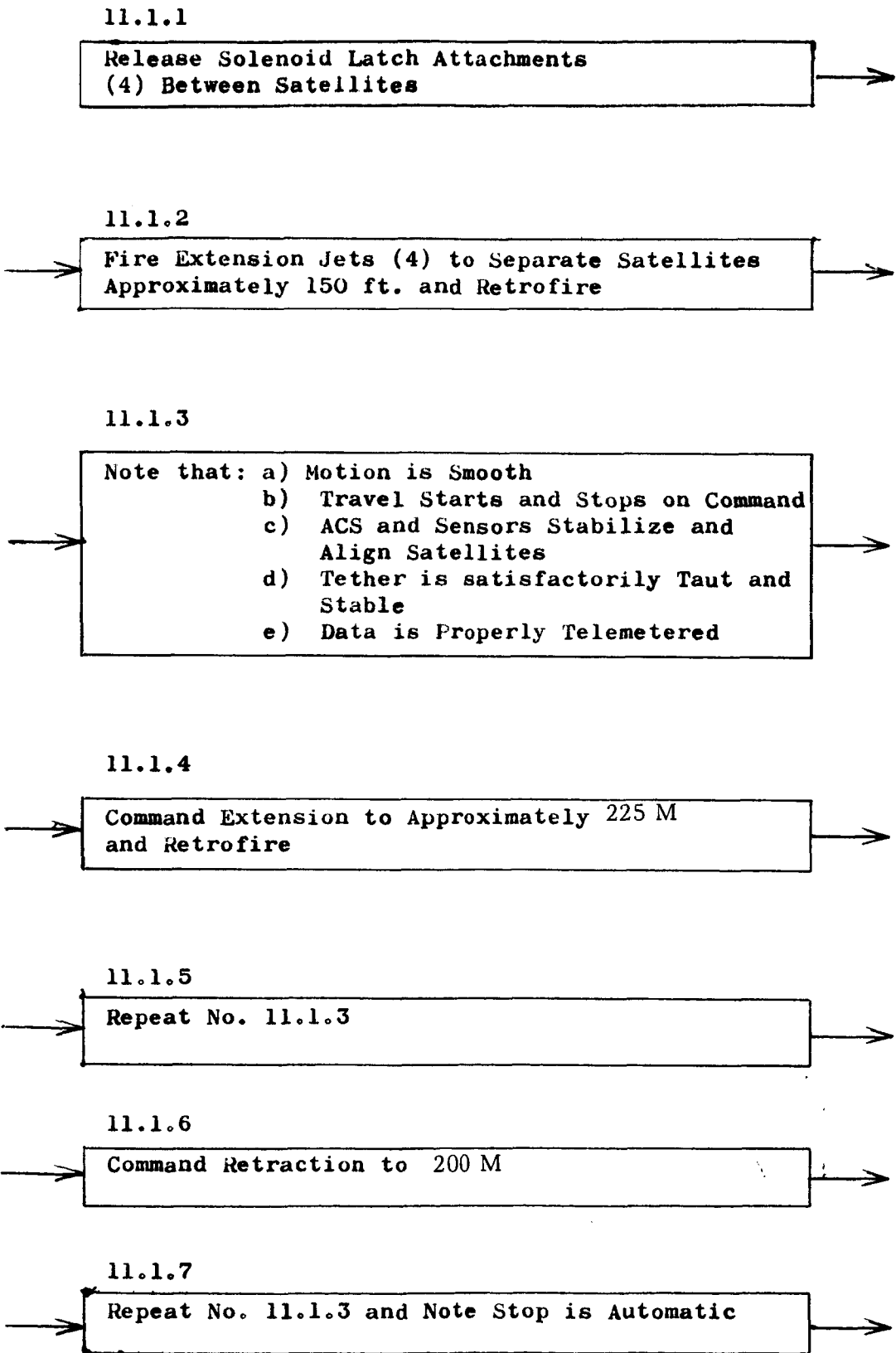
➝ Inspect Antenna for Possible Damage
During Launch or Deployment

Figure I-3. Third Level Functions (Sheet 1 of 5)

11.1 CHECKOUT (PRELIMINARY) TETHER DRIVE SUBSYSTEM

11.1.1

Release Solenoid Latch Attachments
(4) Between Satellites

11.1.2

Fire Extension Jets (4) to Separate Satellites
Approximately 150 ft. and Retrofire

11.1.3

Note that: a) Motion is Smooth
           b) Travel Starts and Stops on Command
           c) ACS and Sensors Stabilize and
               Align Satellites
           d) Tether is satisfactorily Taut and
               Stable
           e) Data is Properly Telemetered

11.1.4

Command Extension to Approximately 225 M
and Retrofire

11.1.5

Repeat No. 11.1.3

11.1.6

Command Retraction to 200 M

11.1.7

Repeat No. 11.1.3 and Note Stop is Automatic

Figure I-3. Third Level Functions (Sheet 2 of 5)

11.3 CHECKOUT BOOM DRIVE SUBSYSTEM

11.3.1

Determine that Clearance Exists for Boom Extension

11.3.2

Command Boom Extension (Approximately 12 in.)

11.3.3

Note that Booms (4) start Moving Smoothly on Command

11.3.4

Command Boom Travel Stop

11.3.5

Note that: a) Booms Stop Promptly
b) Boom Extension Counters Record Corresponding Extensions
c) Boom Drive Tapes and Lead-in Cables are Satisfactorily Taut
d) Extension is properly Telemetered

11.3.6

Command Boom Retraction

11.3.7

Note that: a) Booms Move Smoothly and Uniformly
b) Tapes and Cables Remain Satisfactorily Taut
c) Booms Stop Automatically
d) Extension Counters Record Motion
e) Motion is Properly Telemetered

11.3.8

Command Boom Extension to 7.5M

11.3.9

Repeat No. 11.3.7

11.3.10

Repeat No. 11.3.8 & No. 11.3.7 for 15M & 30M

11.3.11

Command Boom Retraction

11.3.12

Extend Boom to 30M and Proceed to No. 11.4

Figure I-3. Third Level Functions (Sheet 3 of 5)

11.4   CHECKOUT DIPOLE SUBSYSTEM

11.4.1

Extend Booms to 30M

11.4.2

Note the Dipole Length Sensors Readout Zero & Dipole Head Angle Sensors Indicate Stability

11.4.3

Command Dipole Extension for Several Meters

11.4.4

Command Dipole Travel to Stop

11.4.5

Note that Motion is Smooth, Travel Starts & Stops on Command, Dipole Lengths are Properly Telemetered

11.4.6

Command Dipole Retraction

11.4.7

Repeat No. 11.4.5 Except Travel Stops Automatically

11.4.8

Command Dipole Extension to 75M, 150M

11.4.9

Repeat No. 11.4.7 for each Length

11.4.10

Command Dipole Retraction to 75M, 37.5M and Zero

Figure I-3.   Third Level Functions (Sheet 4 of 5)

11.6 CHECKOUT (FINAL) TETHER SUBSYSTEM

11.6.1

Command Extension to 2000M and
Retract to 1000M

11.6.2

Note that: a) Motion is Smooth
           b) Travel Starts on Command and Stops Automatically
              if Required
           c) ACS and Sensors Stabilize and Align Satellites
           d) Tether is Satifactorily Taut and Stable
           e) Data is Properly Telemetered

11.6.3

Command Extension to Approxi-
mately 2100M and Retract to
2000M

11.6.4

Repeat No. 11.6.2

11.6.5

Command Extension to 5000M,
Then Approximately 5100M
and Retract to 5000M

11.6.6

Repeat No. 11.6.2

11.6.7

Command Extension to 10,000 M

11.6.8

Repeat No. 11.6.2

Complete No. 11.7 RF (Final) Checkout & Proceed to No. 12. "Operate"

Figure I-3. Third Level Functions (Sheet 5 of 5)

GROSS FUNCTION	SYSTEM OR COMPONENT FUNCTION	EVENT TIME (MIN)	ELAPSED TIME (MIN)	DISPLAY INDICATION	CREW ACTION OR PARTICIPATION (AT CM SECTION UNLESS OTHERWISE SHOWN)	EVA EQUIPMT.	POSSIBLE EVA HAZARDS	SAFETY OR EMER. PROCD.	FAILURE MODE	FAILURE INDICATION	CREW ACTION OR PARTICIPATION	EVA EQUIPMENT	POSSIBLE EVA HAZARDS	SAFETY OR EMERGENCY PROCEDURES	REMARKS
						NORMAL OPERATION							ABNORMAL OPERATIONS		
Propellant tank refill cont.	2) Command to refill				Receives command from EVA astronaut and opens storage tank valve	Space suit, umbilical, PLSS, tether, batch shoes & will utilize equmt controlled by astronaut to transport replacement items			Valve sticks		Manual manipulation of valve – EVA	Same as equipment mentioned in EVA equipment column to left	Entanglement of umbilical with astronaut & equipment or structure	Adequate stabilization of astronaut to maintain adequate control of umbilical while in EVA to avoid entanglement.	
	3) Tank refills				Shuts off storage valve, drains pressure line & signals to EVA man				Valve sticks		Manual manipulation of valve – EVA				
				Visual	Hose disconnected from tank & returned to SM – by EVA				Hose or fitting bursts under 4000 psi		Replaces hose or component		Injury to astronaut if hose or fitting does burst	To avoid injury to astronaut he should retract to a safe distance after hooking hose up to tank	
Solar cell panel replacement	1) Panel is aligned				Astronaut "A" exits from CM & traverses to panel. Astronaut "B" feeds the new panel from the SM (by means of clothesline) to "A" who aligns & attaches the new panel by inserting & connecting elec. plug. (A) & (B) will return to CM & perform check-out of new panel.	Same as above. Astronaut "B" will have transport equipment for transfer of panels.	If astronaut does not adequately restrain himself at the worksite he could drop the panel.	Astronaut should have adequate restraining device	Attachments jam		Manual manipulation of attachments – EVA	Same as above	Possible entanglement of umbilical with Astronaut and/or structure	Adequate stabilization of Astronaut to maintain adequate control of umbilical while in EVA to avoid entanglement	Old panel remains, new panel added on top
	2) Panel is attached								Attachments jam		Manual manipulation of attachments	Same as above	Same as above	Same as above	
Lead-in harness reel motor replacement	1) Solar panel opened for access				Astronaut "A" exits from CM, traverses to panel & unlatches it. He then unclamps & unplugs motor & transports it by clothesline to astronaut "B" at the SM who replaces it with new motor. "A" then plugs in new motor, reclamps & relatches it. Astronaut "B" in CM commands boom extension & retraction for motor check-out. "A" then closes panel & latches it.	Same as above	Same as above	Same as above	Attachments jam						
Boom drive motor (2) replacement (gears, clutch & solenoid incl)	1) Module removal				IN EVA 1) Astronaut "A" loosens 2 toggles, removes module	Usual plus transport equipment	Normal	Standard	Clamps jam	Manual	Manipulates clamp	Usual	Normal	Standard	
	2) Module exchange				2) Astronaut "A" sends old module to SM. Astronaut "B" returns new module										
	3) New module installed				3) Astronaut "A" locks new module, meshes gear & clamps										
Boom Drive Extension Module replacement (2 drums and gears incl)	1) Inboard drum-pinion relocated to service mounting hole				1) Astronaut "A" aligns new booms in retract position, transports drum-pinion to service mounting, reinserts pin (1 each side)	Usual plus transport equipment	Normal	Standard	Pins jam	Manual	Manipulates pin	Usual	Normal	Standard	
	2) Shaft module removed exchanged and replaced				2) "A" energizes motor to wind up cable onto the driving drums. He then breaks prepared tape splice (2), loosens shaft toggles (2), removes module & sends it to "B" at SM. "B" receives new module & aligns, remeshes gears, clamps it & remakes splice (2)				Motor stalls	Visual	Unclamps shaft & winds manually or replaces motor				
									Splice sticks or won't hold	Manual	Cuts tape & replaces both ends (with drum) manipulates clamp				
	3) Inboard drum-pinion returned				3) "A" replaces drum-pinion inserting it & meshing it with boom rack & pins it (1 each side). Unlocks booms				Clamps jam	Manual	Manipulates clamp				
Boom Drive Extension Tape & Drum replacement	1) Module removal				1) With boom retracted astronaut pulls pins at tape-boom attachment, & at drum axle, & pockets module.	Usual plus replacement	Normal	Standard	Pins jam	Manual	Manipulates pin	Usual	Normal	Standard	
	2) Module replacement				2) He inserts new module, meshes with it & inserts axle pin. He then inserts tape-boom attach pin.										
Boom Drive Retraction Module replacement (shaft, springs, drums, tapes incl)	1) Module removal				1) "A" breaks tape attach at duplex head (or replaces near drum if boom extended), loosens shaft toggles (2), removes module & sends it to "B" at SM.	Usual plus transport equipment	Normal	Standard	Splice sticks	Manual	Cuts tape and replaces both ends	Usual plus tape cutter	Normal	Standard	
	2) Module replacement				2) "A" receives the new module, mounts it on service mounting, extends tape & attaches it (2), relocates module to switch & mesh in place & clamps toggle.	Usual			Clamps jam	Manual	Manipulates clamp	Usual plus replacement			
Tether Replacement	1) Preparation				10,000 M of tether is wound on one reel. "A" re-tains tape on one reel containing transport reel & attaches mounting-attachment to attached satellite drum structure. He then inserts it to inter-satellite docking & tie-down structure.	Usual plus transport equipment & large equipment	Normal plus large equipment	Standard	Reel freezes or motor stalls	Visual	Switches to alternate reel & motor in other satellite and replaces motor	None / Usual plus replacement	None / Normal	Standard	

REFURBISHMENT
TABLE I-1

NORMAL OPERATION

PHASE										**ABNORMAL OPERATIONS**					
GROSS FUNCTION	SYSTEM OR COMPONENT FUNCTION	ELAPSED TIME (MIN)	EVENT TYPE (MIN)	DISPLAY INDICATION	CREW ACTION OR PARTICIPATION (AT CM STATION UNLESS OTHERWISE SHOWN)	EVA EQUIP.	POSSIBLE EVA HAZARDS	SAFETY OR EMER. PROCD.	FAILURE MODE	FAILURE INDICATION	CREW ACTION OR PARTICIPATION	EVA EQUIPMENT	POSSIBLE EVA HAZARDS	SAFETY OR EMERGENCY PROCEDURES	REMARKS
Preparation for Refurbishment	1) Dipoles retracted 2) Booms retracted 3) Tether retracted to one 150M 4) CSM docked to one satellite 5) One satellite docked to the other			CMDP & GDP " " " " " " Visual & CMDP & GDP	Command event " " Astronaut exits CM Astronaut manipulates remote control in EVA	Space suit, umbilical, Dutch shoes, tether, & control unit	Entanglement of umbilical, collision with satellite	Typical EVA plus safe position in relation to advancing satellite	Latch doesn't hold	Local flag	Manually manipulates the latch	Tool kit	Entanglement of umbilical		EVA	CHDP = CM display panel GDP = Ground display panel Retractions are analyzed under checkout Docking is analyzed under deployment
Dipole Module Replacement	1) Old module disconnected and rotated away				Astronaut "A" exits from CM & retracts expanded dipole modules, disconnects 2 pins & an electrical connection. He rotates expanded module to one side.	Space suit, umbilical, Dutch shoes, tether, component, & transport clotheslining & equipment	Entanglement of umbilical. If astronaut does not adequately restrain himself at the work site he could injure himself while installing it.	Astronaut should have adequate restraining device to preclude any accidents	1) Pins or plug jam	Visual and manual	Pries pin or plug with equivalent of large screw driver	Tool kit	Entanglement of umbilical		Typical EVA	Old modules held out by hinge spring
	2) New module transported from SM				Astronaut "A" installs new module by inserting 3 pins & connecting an electrical harness for each module. Upon completion of tank astronaut "A" & (B) return to the CSM.											
	3) New module installed				Astronaut "B" with use of clotheslining transport equipment feeds the new module from the SM to Astronaut "A".			3) Holes damaged		Select undamaged hole or receptacle					Extra hole and receptacle provided
Individual Solenoid/nozzle unit replacement	1) Old unit disconnected and removed				Astronaut "A" exits from CM & traverses to the expended solenoid/nozzle unit. He unclamps the toggle clamp, disconnects quick-disconnect & electrical harness & removes expended unit.	same as above	same as above	same as above	Jam or break in attachments	Visual and manual	Replace jet or entire module	Tool kit and replacement	Entanglement of umbilical		Typical EVA	Replacement of entire module consisting of 4 solenoid/nozzle and one common regulator recommended at time of major refurbishment. See below
	2) Old unit exchanged for new				Astronaut "B" exchanges new unit for old by means of transport equipment											
	3) New unit installed				Astronaut "A" installs new unit by clamping with toggle clamp, plugging into propellant unit & connecting the electrical harness. Upon completion of tank, Astronauts "A" & "B" will return to CSM.											
Solenoid/Nozzle Assembly Replacement Module Installation	1) Replacement module transported to worksite from CSM				IN EVA 1) Astronaut (B) transports module from CSM by transport equipment	Space suit, umbilical, Dutch shoes & clothesline transport equipment	Module could drift leading to injury to astronaut or damage to the structure	Astronaut should be adequately restrained & have complete freedom of both hands	Jam or break jam	Visual & manual	Frees element or replaces module	Tool kit	Entanglement of umbilical with astronaut and/or structure		Adequate stabilization of astronaut to maintain adequate control of umbilical while traversing to and occupied at worksite	Replacement module is added on top of the expended assembly. The expended assembly can be replaced by a new identical assembly (not a replacement module) as an alternate method.
	2) K/M is aligned and mounted				2) Astronaut (A) pulls 4 pins, attach pins, align & mounts replacement module	same as above		same as above	Jam or break		Frees element or replaces module					
	3) K/M is connected				3) Astronaut (A) inserts attach pins and quick disconnect											Electrical switching is automatic
Propellant tank replacement	1) Solenoid/nozzle assembly rotated out of the way.				IN EVA 1) Astronaut (A) pulls 2 pins and rotates S/N	same as above except extra new part & new tank supplied by astronaut (B) from CSM	The tank could drift leading to injury to astronaut & structure or damage to the structure	same as above	Jam or break	Visual & manual	Frees element or replaces module	Tool kit	Entanglement of umbilical		Same as above	Assembly can be removed completely as above but removal is not necessary
	2) Old tank exchanged for new				2) Astronaut (A) pulls 4 pins, removes the tank & exchanges it for new tank				Jam or break		Frees element or replaces module					
	3) New tank installed				3) Astronaut (A) mounts and puts new tank				Jam or break		Frees element or replaces module					
	4) S/N replaced and reconnected				4) Astronaut (A) rotates S/N into place and attaches and connects it				Jam or break		Frees element or replaces module					
Propellant tank refill	1) Resupply hose in tank, inserted and attached				IN EVA 1) Astronaut (A) receives hose from Astronaut (B) at CSM, inserts nozzle of hose & nozzle, then withdraws to safe position before signaling.	same as above except new part	Hose could get loose from astronaut to injure him or do damage to structure	Astronaut should have adequate control of hose & should be adequately restrained at all times	Reel jam Nozzle jam	Visual & manual	Frees reel	Tool kit	Entanglement of umbilical		Same as above	Tank can be pre-pressurized or can be filled as follows

REFURBISHMENT
TABLE I-1   PHASE _____

				NORMAL OPERATION						ABNORMAL OPERATIONS				
GROSS FUNCTION	SYSTEM OR COMPONENT FUNCTION	EVENT TIME (MIN)	ELAPSED TIME (MIN)	DISPLAY INDICATION	CREW ACTION OR PARTICIPATION (AT CM STATION UNLESS OTHERWISE SHOWN)	EVA EQUIP.	POSSIBLE EVA HAZARDS	SAFETY OR EMER.PROCD.	FAILURE MODE	FAILURE INDICATION	CREW ACTION OR PARTICIPATION	EVA EQUIPMENT	POSSIBLE EVA HAZARDS	SAFETY OR EMERGENCY PROCEDURES	REMARKS
Tether Replacement (continued)	2) Preliminary tether splices				2) "A" breaks prepared tether splice near empty reel & joins splice of new tether. He then connects the loose end of the old tether to empty portion of transfer reel.				Splice won't break or won't hold	Manual	Uses alternate splice	Usual	Normal	Standard	A redundant duplication is provided at each splice
	3) Tether transfer				3) The empty satellite reel is then rotated and filled from the transfer reel, winding old tether onto that reel										
	4) Final tether splice				4) "A" disconnects old tether at the new empty satellite reel and splices in new tether.										
	5) Clean up				5) Transfer reel is disconnected and transported back to SM.										
Tether Drive Replacement	1) Module removal				1) "A" loosens toggle motor & speed reducer & sends it to SM	Usual plus transport equipment	Normal	Standard	Clamp jams	Manual	Manipulates clamp	Usual	None	Standard	
	2) Module replacement				2) "A" receives new module, aligns, meshes and clamps it in place										

APPENDIX II

NASA FORM 1346

Contained herein is the completed NASA form 1346, which summarizes the proposed crossed-H interferometer antenna. Although the actual scientific experiments which will be flown are not yet determined, a "hypothetical observation program" was conceived and utilized as a design guide. Presentation of this model program is included in the form for reference only.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

EXPERIMENT PROPOSAL

FOR

MANNED SPACE FLIGHT

TITLE___CROSSED-H INTERFEROMETER_____

(Confine to total of 30 letters, numerals, spaces, punctuation marks, etc.)

EXPERIMENT NUMBER

PRINCIPAL INVESTIGATOR_____
                              (Signature)   *(Date)*

PRINCIPAL ADMINISTRATOR_____
                              (Signature)   *(Date)*

NASA FORM 1346   JAN 67 REPLACES NASA FORMS 1138, 1138A, B, AND C, WHICH ARE OBSOLETE.

THIS DOCUMENT PROVIDES THE FORMAT TO BE FOLLOWED BY THE INVESTIGATOR OR PROPOSING INSTITUTION, WITH SUPPORT AS REQUIRED FROM THE NASA SPONSORING PROGRAM OFFICE, WHEN SUBMITTING PROPOSED EXPERIMENTS FOR MANNED SPACE FLIGHT TO NASA FOR REVIEW AND ACCEPTANCE. INFORMATION REQUESTED SHOULD BE COMPLETED AS ACCURATELY AND WITH AS MUCH DE- TAIL AS POSSIBLE, SINCE THIS DOCUMENT WILL PROVIDE THE PRI- MARY DATA FOR AN EVALUATION OF EXPERIMENT MERIT AND DETERMINATION OF EXPERIMENT COMPATIBILITY TO A MANNED SPACE MISSION. ALSO, THE TECHNICAL ENGINEERING, AND OPERA- TIONAL INFORMATION (SECTIONS II, III, AND IV) CONTAINED IN THIS PROPOSAL, WHEN UPDATED, WILL CONSTITUTE THE EXPERIMENT DESCRIPTIVE INFORMATION PORTION OF THE EXPERIMENT IMPLE- MENTATION PLAN, TO BE PREPARED BY A NASA CENTER BEFORE FINAL APPROVAL OF THE EXPERIMENT CAN BE AUTHORIZED.

FOR MONITORING PURPOSES, PLEASE NUMBER THE PAGES OF YOUR PROPOSAL PROGRESSIVELY FROM THE FIRST TO THE LAST AS FOLLOWS: PAGE 1 OF N PAGES, PAGE 2 OF N PAGES . . ., PAGE N OF N PAGES.

SECTION I - ADMINISTRATIVE /BIOGRAPHICAL

1.	APPLICANT INSTITUTION	

Name of Applicant Institution	Type of Institution ☐ Government ☐ University	☐ Non-Profit ☐ Industrial ☐ Other

Address	Telephone

Name of Principal Administrator Responsible for Experiment	Title

2.	PRINCIPAL INVESTIGATOR	

Name of Principal Investigator	Title

Mailing Address	Telephone

Biographical Sketch:
Brief summary of education, experience and professional qualifications.

3.	OTHER INVESTIGATORS	
NAMES	MAILING ADDRESSES	TITLES OR POSITIONS

4. **RESEARCH SUPPORT**

List all other funded and proposed research support of the principal investigator. Include support for this project received from own organization. Amounts shown should reflect total funds awarded over the entire grant periods indicated in the final column.

a. NATIONAL AERONAUTICS AND SPACE ADMINISTRATION SUPPORT

GRANT/CONTRACT NUMBER	TITLE OF PROJECT	APPROXIMATE PERCENT TIME/ EFFORT ON PROJECT	TOTAL AMOUNT ($)	TOTAL PERIOD OF SUPPORT WITH DATES

b. ALL OTHER RESEARCH SUPPORT

SOURCE AND GRANT/ CONTRACT NUMBER (If designated)	TITLE OF PROJECT	APPROXIMATE PERCENT TIME/ EFFORT ON PROJECT	TOTAL AMOUNT ($)	TOTAL PERIOD OF SUPPORT WITH DATES

NASA FORM 1346 Jan 67   2

5. PRINCIPAL INVESTIGATOR'S ROLE IN RELATION TO THIS EXPERIMENT
(Include percent of time to be spent on this project)

6. RESPONSIBILITIES OF OTHER KEY PERSONNEL
(Include percent of time to be spent on this project)

SECTION II – TECHNICAL INFORMATION

1. OBJECTIVES

The flight objectives of the crossed-H interferometer are to:

a. Evaluate the role of man in the deployment, assembly, alignment, main-
tenance, and repair of large structures in space. Through the use of photog-
raphy, the astronaut's abilities to accomplish the assigned tasks may be
recorded. Biomedical sensors will provide additional data on man's capa-
bilities in the performance of such tasks as the transportation, handling, and
replacement of various components.

b. Evaluate the performance and behavior of large structures in space. Through
the media of thermal sensors, strain gauges, and photography, the structural
behavior may be recorded. Operation of the deployment and adjustment mech-
anisms, thermal and dynamic distortions, and structural/mechanical degrada-
tion in the space environment are of interest in the advancement of space
structures technology.

c. Provide a useful long wave radio astronomy antenna capable of accomplishing
a wide variety of radio observations. Although the actual scientific experi-
ments which are proposed to fly on the antenna are yet to be determined, a
"hypothetical scientific observation program" was conceived and utilized as
the design goal throughout the conceptual phase. Inherent flexibility has been
built into the hypothetical scientific observation program (and therefore into
the design of the antenna) and it is anticipated that the basic configuration will
handle most of the long wave radio astronomy observations which are scien-
tifically desirable in the time frame under consideration. A description of
this program is as follows:

(1) Survey the VLF sky radiation over the entire sky, with good resolution,
including spectral and polarization measurements.

(2) Survey VLF discrete radio sources, with good resolution, including
spectral and polarization measurements.

(3) Obtain spectral and polarization measurements of the sun, with good
temporal resolution.

(4) Obtain VLF observations of Jupiter, and possibly other planetary sources,
with good temporal resolution.

2. SIGNIFICANCE

Man's role in the initial deployment and checkout, malfunction repair, and sched-
uled refurbishment of the crossed-H interferometer will expand and advance man's
capabilities in support of future large structures in space.

NASA FORM 1346   Jan 67   5

The deployment and operation of this large structure in orbit will contribute to technological advancements spanning three distinct types of structures of concern to future space systems: tethers, trusses (booms), and extendible tubular elements (dipoles).

The interferometer is a space structure that will, in the future, greatly expand man's capability to measure and understand the phenomena exhibited by the bodies of the solar system and the electromagnetic radiation reaching us from the most distant stars. The crossed-H interferometer will contribute significantly to radio-astronomy science by making measurements that can only be made from space, providing greater sensitivity than generally available, and greater resolution for mapping the radio sources.

3. DISCIPLINARY RELATIONSHIP

    a. Related Work

        A major objective is to verify the role of the astronaut in erecting and maintaining the operation of a large orbiting structure. The proposed role of man is an extension of previously demonstrated capabilities of rendezvousing, docking, observing activities on the other spacecraft, photography, and EVA. The EVA includes the demonstrated capabilities of the astronauts to move about the experiment, using handholds, tethers, and other EVA aids. Extension of these capabilities, which may be demonstrated in orbit before the crossed-H interferometer launch, includes the capability to replace and repair structure assemblies, spacecraft subsystem components, and use new EVA aids such as the "clothesline supply."

        One of the prime objectives of the crossed-H interferometer is the advancement of the technologies of large, orbiting structures. One technology area is the tether dynamics. Except for the Gemini/Agena tether experiment, and developmental programs using tethers for gravity-gradient stabilization, related work on tether dynamics has been limited to studies and simulation. The extendible boom is an extension of current technology to the space environment. Much is still to be learned about thermal and dynamic distortions through instrument action by thermal and strain gages. The bi-metallic mesh extendible/retractable tubes have been developed; however, there has not been a demonstration of their capabilities in space.

        Related work to the scientific objectives has been limited almost entirely to radio-source surveys using ground-based antenna systems. Dipole elements have been orbited by Haddock, Huguenin, and others, and the RAE satellite, soon to be orbited, has VLF capability with increased resolution. The requirements of the scientific community will not be fully met until the large structure of the crossed-H interferometer is orbited.

b. Present Development in Field

As explained above, the development of the crossed-H interferometer is based on extensions of current technologies into larger structures, lighter structures, and space applications.

4. EXPERIMENT APPROACH

a. Experiment Concept

The interferometer experiment system consists of two satellites which are stabilized in the earth's gravitational field through the utilization of an adjustable length tether. (Ref: Figure II-1) The two satellites are geometrically identical and vary only within the electronic receiving and transmitting networks contained within the centerbodies. Each satellite consists of five major components: the center body, two extendible boom assemblies and two dipole head assemblies. Location of the major features of each satellite is illustrated in Figure II-2.

In order to achieve the broad performance range exhibited by the antenna system, the basic geometry of the antenna has been made adjustable. The tether system is adjustable from 0 to 10,000 meters in length. The telescoping booms which support the dipole heads are adjustable from 3 meters to an over-all maximum of 30 meters, while the individual dipole lengths are variable from 6 meters to 75 meters.

The entire antenna array may be retracted to the launch configuration for gross EVA maintenance and refurbishment functions.

b. Experiment Procedure (Ref: Figure II-10)

The deployment sequence in orbit is as follows:

(1) The CSM is separated from the launch vehicle.

(2) It turns and docks to the payload support structure.

(3) The experiment payload is separated from the launch vehicle and moved clear using the CSM attitude control.

(4) Pre-deployment subsystem checks are performed by the crew.

(5) The interferometer experiment is separated from the CSM.

(6) The tether system is now activated to separate the satellites to 250 meters and retracted to 200 meters as a check on the tether mechanism. The boom assemblies and dipole elements are also deployed. Should a malfunction be indicated, the CM docks to the affected satellite for EVA

Figure II-1. Crossed-H Interferometer Experiment

1. MAIN BODY ASSEMBLY - ACS SYSTEM
   TETHER REEL AND TENSION SENSORS
   RECIEVING AND TRANSMITTING ELECTRONICS
   GROUND DATA LINK ELECTRONICS
   STAR TRACKING SYSTEM
   GUIDANCE AND CONTROL ELECTRONICS
   BATTERY AND CHARGING SYSTEM
   BOOM ACTUATION MOTOR
   STRUCTURAL SENSORS AND TELEMETRY

2. TRIANGULAR TELESCOPING BOOM - 1 FIXED, 7 MOVING SECTIONS.

3. DIPOLE HEAD ASSEMBLY

4. DEPLOYABLE MESH DIPOLE UNIT (8)

5. TETHER TAPE - ADJUSTABLE 0 - 10,000 METERS

6. ATTITUDE CONTROL JET MODULE (9)

7. ATTITUDE CONTROL PROPELLANT

8. SOLAR CELL PANELS - 115 SQ.FT.

9. ATTITUDE CONTROL JET MODULE (2)

10. CSM SATELLITE DOCKING RING

11. INTER-SATELLITE DOCKING STRUCTURE

12. STAR TRACKER OPTICS (2)

13. EVA RESTRAINT RAILS

Figure II-2.   General Arrangement

5

repair. After completion of the mechanical test, the dipoles are retracted and the antenna deployed to the full 10,000 meter separation.

(7) The experiment spacecraft is deployed to its first operational configuration by remote control of the astronauts.

The CSM remains in the vicinity of the experiment during the systems checkout phase and the initial 14 to 28 days of operation so that any early malfunctions may be corrected by the astronaut prior to his departure from orbit.

During the orbital operational life of 2 years, the tether length connecting the two satellite bodies is varied from 62.5 meters to 10,000 meters to cover the desired frequency spectrum. During the two years of operation, 427 days are allocated for sky and galactic mapping, and the remainder of the time is available for discrete source observation and maneuvering.

c. Measurements

Biomedical data measurements required during astronaut EVA will include EKG, impendance pneumograph (respriation rate/volume), and body temperature. Time/task analysis will require flight crew observation and photographic recording of the EVA.

Structural performance evaluation will require that deployment rate, positions, and alignment be measured. This will be correlated with measurements of stress/strain and temperature at critical points on the structure.

A preliminary measurement program for the scientific observations is as follows:

(1) Scan celestial sphere at 0.5 to 10.0 MHz.

    (a) Sweep equatorial plane looking alternately E and W.

    (b) Reorient to the NE by 5-deg steps looking alternately NE and SW for 18 steps (90 deg).

(2) Scan celestial sphere at 3.5, 5, 7, and 10 MHz frequencies. Set configuration of the antenna system to optimum for the specific frequency for each of four scans.

(3) Acquire and lock on selected targets, maintaining a fixed-configuration traverse to various targets, then change configuration for different frequency and repeat.

The range of numerical values expected varies from the signal strength of the strongest radio sources to the maximum sensitivity of the interferometer antenna/receiver system. Data storage is not provided in the spacecraft, and data is retransmitted by the experiment elemetry system as required

for recording on the ground. There is no requirement for telemetry capacity in the CSM to support the orbital operations of the crossed-H interferometer.

d. Analysis and Interpretation of Data

Data analysis and interpretation is the responsibility of the principal investigator and will be determined at a later date.

e. Prime Obstacles and Uncertainties in the Experiment Approach

Design of the crossed-H interferometer experiment is based on current state-of-the-art in development of materials and construction details. As such, there are no direct obstacles to the accomplishment of this experiment. Continued advancement in associated technologies will accrue favorably to this experiment.

Two areas of technological advancement could significantly contribute to the technical credibility of the proposed interferometer, both of which may be satisfied by related satellite programs. The dynamic behavior of tethered, gravity gradient stabilized satellites in synchronous orbit could be verified by other systems placed in orbit prior to the crossed-H interferometer, such as the proposed tethered orbiting interferometer (TOI) studied by the Applied Physics Laboratory and R. Stone of Goddard Space Flight Center. A second area of concern is the dynamic/thermal behavior of long extendible tubular elements (for dipole application in the case of the crossed-H). Here again, GSFC is conducting radio astronomy experiment tests of benefit to this proposed antenna.

f. Astronaut Participation

The astronaut's role in the experiment is to serve in the areas of deployment, observation, checkout, malfunction repair, and gross refurbishment. The function of man has been analyzed and the results reflected in the basic design of the experiment to provide the greatest possible assurance of mission success.

The basic satellite systems and component parts have been analyzed on a matrix of expected failure rates, astronaut capabilities, hazards, and system costs. This analysis serves as the basis by which the operational modes of the systems are selected. Component part redundancy and replacement by EVA were weighed for system effectiveness. Using this approach, man's activities have been designed into the mission only when justified on the basis of increased probability of mission success.

Man's activities during the experiment sequence are as follows:

(1) Deployment monitoring.

(2) Post deployment equipment checkout.

(3) Replacement of malfunctioning modular equipment in the following sample areas:

    (a) Solar cells

    (b) Actuator motors

    (c) Dipole units

    (d) ACS modules

    (f) Batteries

    (g) Radiometry components

    (h) Guidance system components

(4) Gross refurbishment at the end of one-year operation is as follows:

    (a) Replenishment of ACS fuel supply

    (b) Replacement of dipole-head assemblies

    (c) Replacement of marginal equipment as listed in (3)

Replaced faulty equipment will not be removed from the satellite prior to the gross refurbishment operation. Faulty equipment will be either swung aside to provide replacement space or left in place and overlaid by the new equipment module. All EVA-oriented equipment is provided with quick disconnect or snap-on type mountings.

5. BASELINE OR CONTROL DATA

Support activities that are recommended to augment the crossed-H interferometer experiment are the three prerequisite orbital experiments:

a. Clothesline Supply

This is an orbital experiment in which a remotely-located EVA astronaut is supplied with tools and materials that are then returned to the CSM through the use of a "clothesline" EVA aid.

b. Astronaut Locomotion Loads

This experiment is recommended to determine the actual loads on the typical large space structures resulting from astronaut locomotion in contact with the structure, in the space environment.

c. Boom Deflection

An orbital experiment is recommended to measure the static and dynamic characteristics of tubular dipole boom materials in the thermal and dynamic space environments.

The design of the crossed-H interferometer experiment is based on current state of the art in development of materials and of construction details. While not specifically required, research in the following areas will contribute to performance, cost, or schedule improvements in the development of the crossed-H interferometer.

a. Instrumentation for measuring boom deflection in space.

b. Analysis of effects of distortion on the long dipoles.

c. Fabrication techniques for thin-walled seamless beryllium and titanium tubing.

d. Techniques to reduce manufacturing tolerances on screen-boom tubing.

e. Dynamic analysis of complete space structures.

f. Control system analysis for large, flexible, space structures.

g. Evaluation of EVA capabilities, and developments, and predictions for future capabilities.

SECTION III – ENGINEERING INFORMATION

1. EQUIPMENT DESCRIPTION

a. Experiment Hardware

The experiment hardware comprises the large space structure of the interferometer antenna, the satellite structures and support structre, and the satellite subsystems:

(1) Radiometry receiving system

(2) Data transmission and telemetry system

(3) Data processing system

(4) Navigation and attitude control system

(5) Power system

A block diagram of the crossed-H equipment is given in Figure II-3.

(1) Radiometry

Principal elements of the radiometry receiving subsystem are the dipole antennas, the matching and phasing networks, the detectors and correlators, and the timing and phasing circuits. Other elements are the amplifiers, the filters, the upper-to-lower satellite data relay. Two sets of crossed dipoles are located on each satellite at the boom ends. They form end-fire arrays that are designed to provide a cardioid-like radiation pattern, with enhanced reception in one hemisphere to give directivity to the system. Dipole lengths and spacings are:

Frequency (MHz)	Dipole Length L (meters)	Dipole Spacing d (meters)	$\dfrac{L}{\lambda}$	$\dfrac{d}{\lambda}$
0.5 2.5	150	30	1/4 5/4	1/20 1/4
2.5 5	75	15	5/8 5/4	1/8 1/4
5 10	37.5	7.5	5/8 5/4	1/8 1/4

For operation of the interferometer in the mapping mode the configuration of the dipoles and circuits in each satellite are kept identical. For the observation of time-varying sources during which the interferometer function is not used, the configuration of the two satellites is made different so that measurements in different frequency bands can be made simultaneously.

Figure II-3. Simplified Equipment Block Diagram

(2) Data Transmission and Telemetry

The functioning of the interferometer requires the exchange of a considerable amount of data between the two satellites as well as between the lower satellite and the ground, as follows:

(a) Ground-to-Satellite . . . . Orientation commands
Adjustment commands
Tuning commands
Time reference signals

(b) Lower Satellite-to-Upper Satellite . . . . . . Command signals
Timing signals

(c) Upper Satellite-to-Lower Satellite . . . . . . Status information
Configuration information
Orientation information
Raw radiometry data

Data rates between the two satellites must be high to satisfy the timing requirements and also to accommodate the raw experimental data. In these links the data rate is 5×10^6 bits per second. For the earth-lower satellite links the rates are much lower: 10 kilobits per second. This lower data rate is possible because of the on-board processing of the radiometry data.

These systems are pending definition of detailed equipment requirements by the principal investigator(s). When defined, the design ground rule of compatibility with the MSFN and DSIF ground stations will be applied.

(3) Data Processing

An on-board digital computer is provided to process the raw radiometer data.

(4) Navigation and Attitude Control

This subsystem uses star trackers in the upper satellite, and a star tracker – horizon scanner combination in the lower satellite to determine the orientation of each satellite. An on-board digital computer is then used to compute the commands required to achieve the proper orientation. A cold-gas (nitrogen) reaction system is used to perform the required maneuvers. Inputs to the subsystem consist of orbital parameters which are determined on the ground and transmitted to the spacecraft, and orientation data from the on-board sensors. This data is fed into the computer in the lower satellite which then computes duration of reaction jet operation, rate and acceleration.

(5) Power

The power subsystem is a conventional solar cell/battery system. The solar cells are body-mounted on both the upper and the lower satellites. Flat panels, matching the satellite facets, are used to achieve an omnidirectional system which yields an average of 307 watts. Energy storage is based on a high power operation during the dark period of 72 minutes. Limiting the depth of drain to 40 per cent results in a 900 watt-hour battery. Twenty-seven silver-cadmium cells at a rating of 83 ampere-hours are used to provide 28 vdc for the system.

b. Required Equipment	c. State of Definition
1. Flight Article (FA)	Conceptual Design
2. Engineering Model (EM)	Conceptual Design
3. Engineering Subassembly Test Articles (SA-1 to -11)	Conceptual Designs
4. Training Articles Neutral Buoyancy Tank (TA-1)	Available
Sky Mapping Console Simulator (TA-2)	Conceptual Design
Orbital Mission Simulator (TA-3)	Conceptual Design
5. Manufacturing Tooling Fixtures	Designs
6. Test Fixtures	Available Components
7. Handling Equipment	Available Components
8. Launch Control Panel	Conceptual Design

2. ENVELOPE

The crossed-H interferometer experiment has six major assembly types. The assemblies, dimensions, and configurations are shown in sketches as follows:

Assembly 1, Upper Satellite Body	Figure II-4
Assembly 2, Lower Satellite Body	
Assembly 3, Boom (4)	Figure II-5
Assembly 4, Dipole Head Module (4)	Figure II-6
Assembly 5, Tether	
Assembly 6, Support Structure	Figure II-7

Note: General Configuration and Dimensions of
Lower Satellite Body are Same as Shown

72.42 IN.

72.42 IN.

72.42 IN.

SOLAR PANELS (15)
30 IN. × 30 IN.
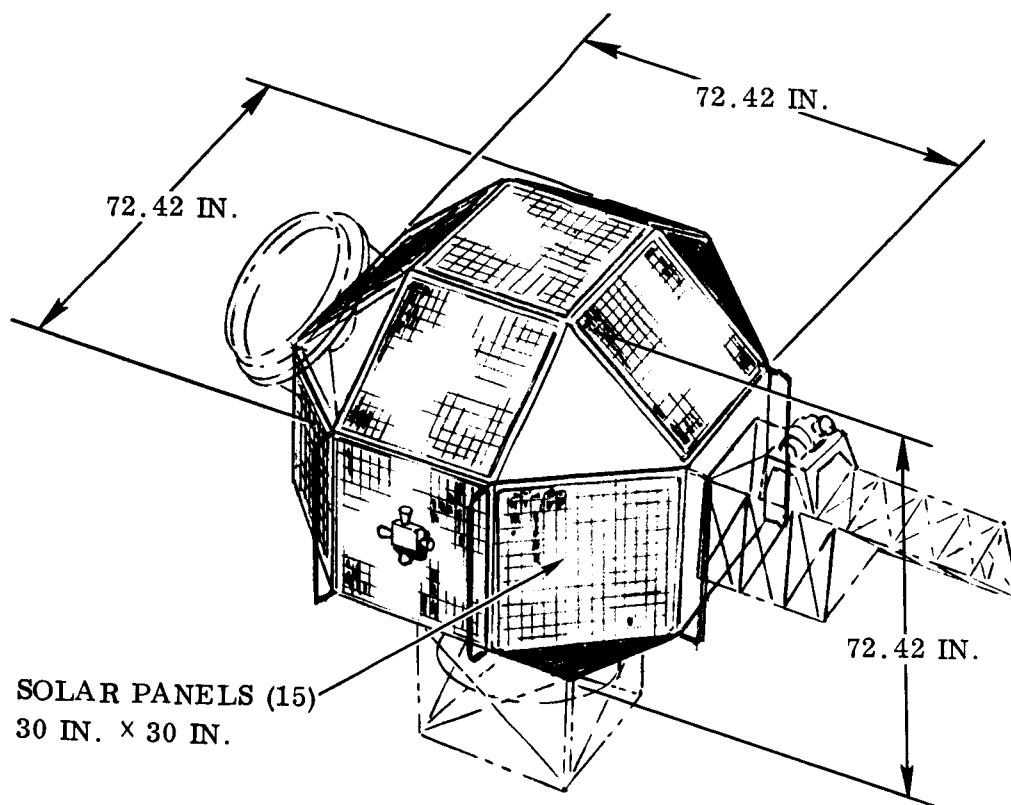
Figure II-4. Upper Satellite Body Assembly

42.5 IN. STOWED

46.3 FT EXTENDED
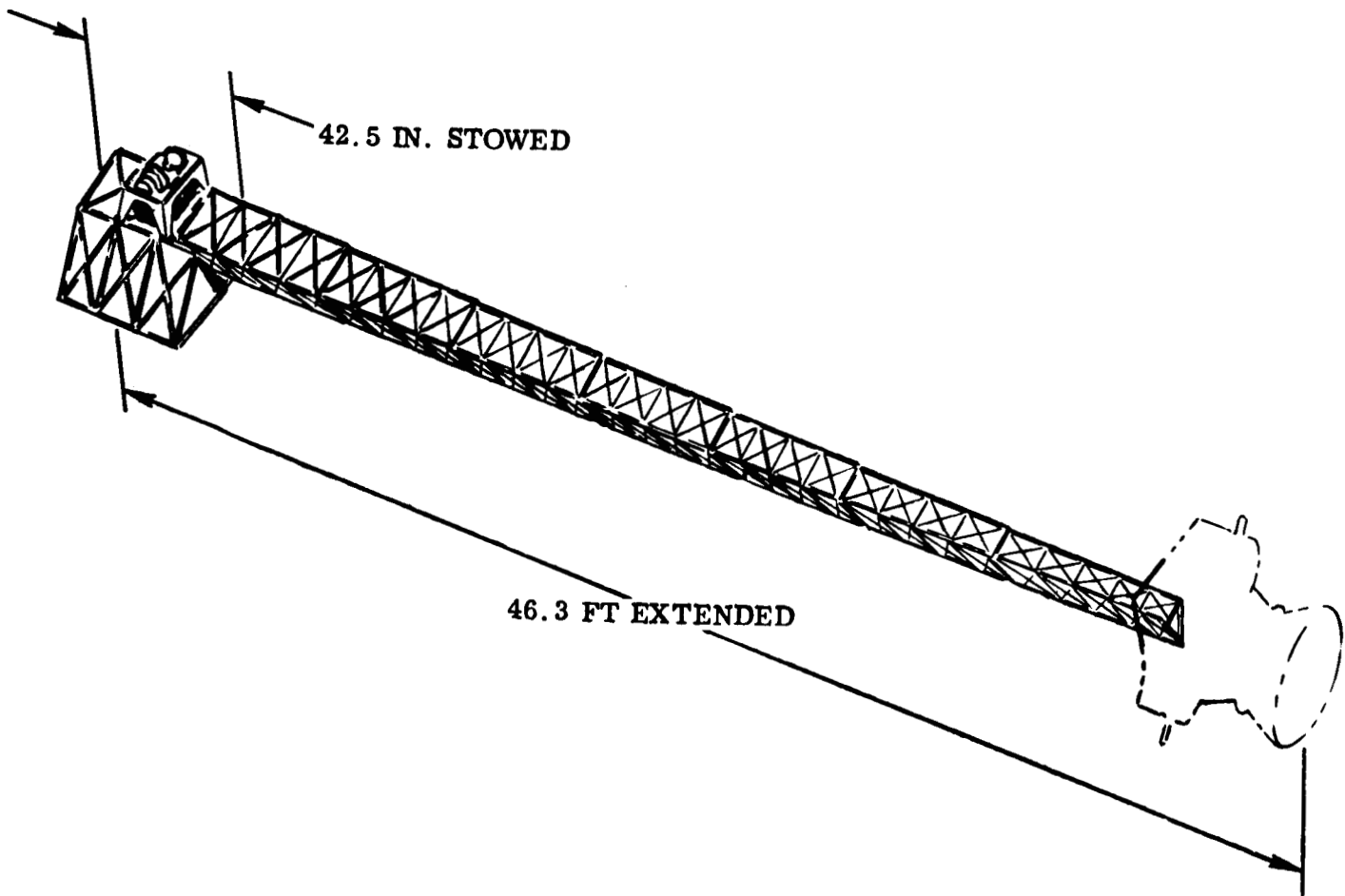
Figure II-5. Boom Assembly

24 IN.

24 IN.

29 IN.
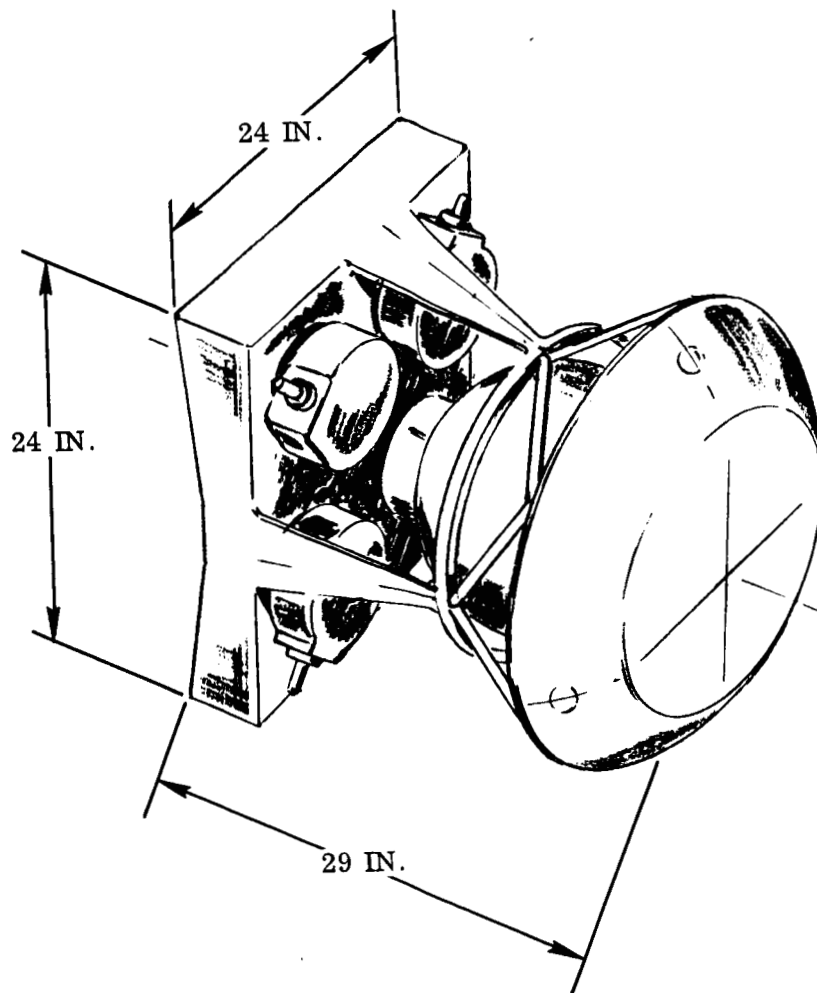
Figure II-6. Dipole Heat Module Assembly
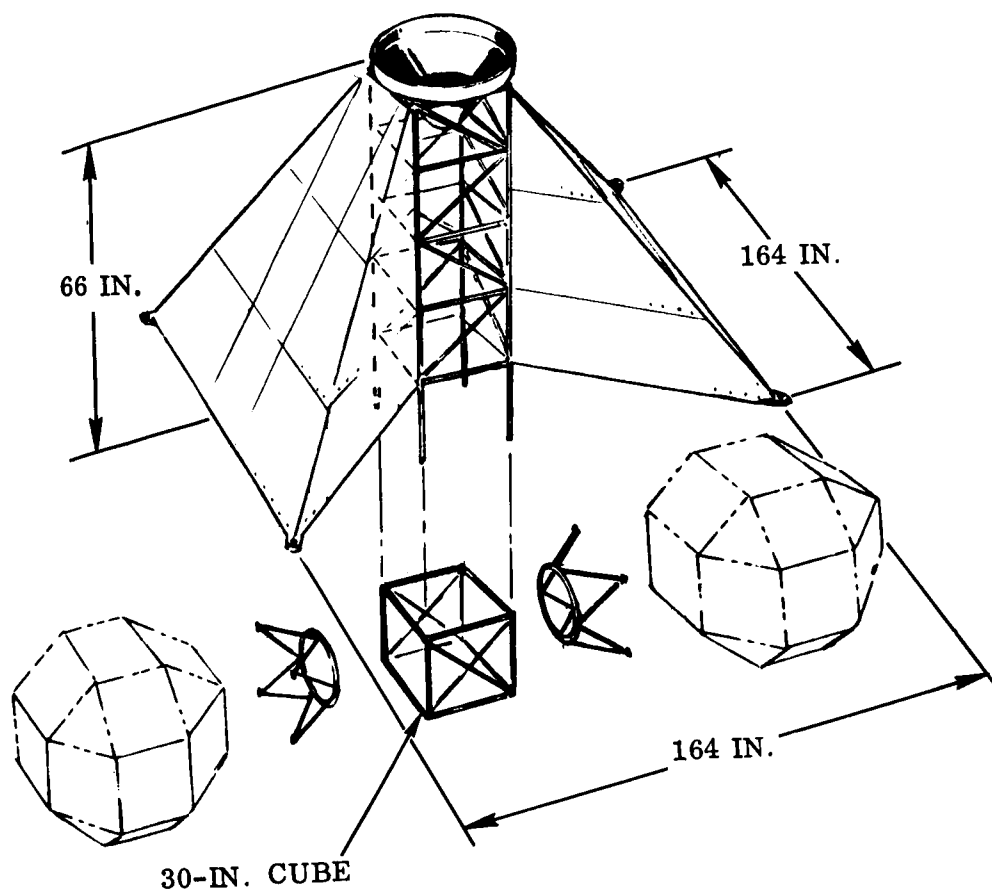
Figure II-7. CSM Dock and Support Structure

3. WEIGHT AND SIZE

Equipment Item	Weight (lb)	Volume (cu ft) Stored	Operation	Dimensions (in.) Stored	Operation	Shape Stored	Operation
1. Upper Satellite Body	932.0	140	140	72.5 × 72.5 × 72.5		26–sided Spheroid	
2. Lower Satellite Body	1033.5	140	140	72.5 × 72.5 × 72.5		26–sided Spheroid	
3. Booms (4)	926.2	48	Variable		Variable	Triangular Boom	
4. Dipole Head Modules (4)	807.0	38	38	24 × 24 × 29		Roughly Cubic	
5. Tether	123.6	(NA)	(NA)	(10,000 Meters)		(Tape)	
6. Support Structure	279.5	(NA)	(NA)	164 × 164 × 96		Pyramid	(NA)
TOTAL	4101.8						

4. POWER

(The spacecraft has independent power sources and therefore does not require power from the CSM.)

The average power requirement available in the satellite is 307 watts.

5. SPACECRAFT INTERFACE REQUIREMENTS

a. Required or Desired Location

The crossed-H interferometer is configured for launch within the LEM Adapter. Although the desired location, within this volume, is shown in Figure II-8, the antenna can be packaged nearly anywhere within the LEM adapter and appropriate support can be made available.

b. Mounting Requirements

The pyramidal support structure for the crossed-H is tailored to the four LEM attach points at Station 3341. Interfaces at these points are mechanically and functionally equivalent to those at the LEM/LEM adapter interface.

c. Subsystem Support Requirements

Subsystem support requirements are limited to the prelaunch phase, except for specific service functions provided through EVA. Subsystems having support requirements on the CSM and Saturn launch vehicle are:

Subsystem	Requirement
Radiometry Receiving System	Landline Checkout
Data Transmission and Telemetry	RF Data Link Checkout
Data Processing	Landline Checkout
Navigation and Attitude Control	Landline Checkout
Power	External Power and Landline Checkout

d. Control Requirements

The CSM is used to control the combined spacecraft and support structure payload in the process of separating it from the launch vehicle. Interface here is a mechanical docking ring. Docking rings are also provided on each satellite body for maintenance and repair.
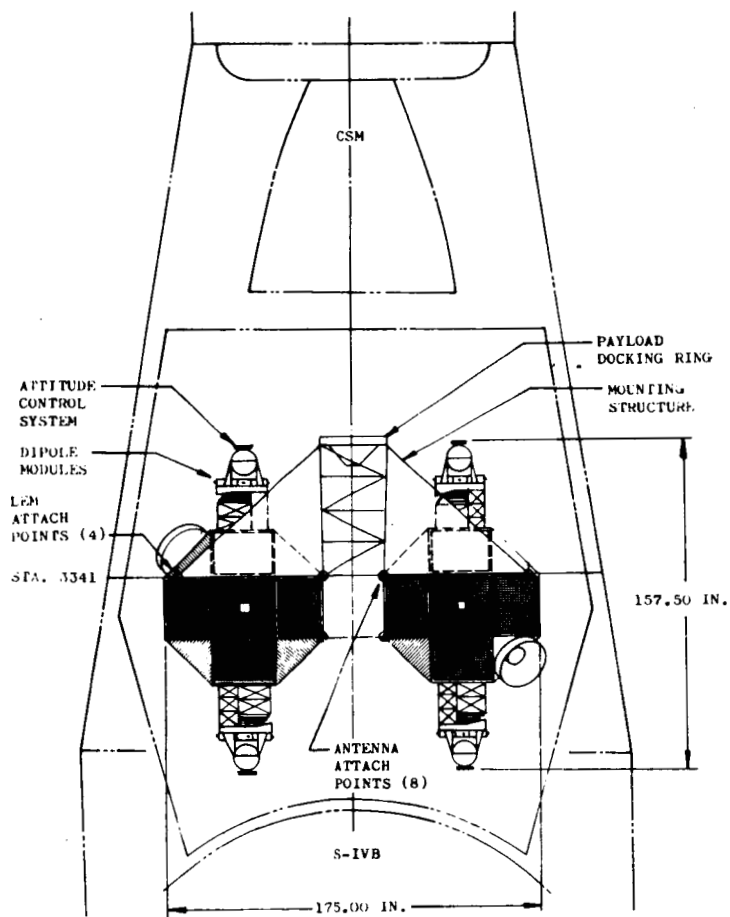
Figure 8. Desired Experiment Location

NOTES:

    (1) Upper satellite body.

    (2) Lower satellite body.

    (3) Booms.

    (4) Dipole head modules.

    (5) Tether.

    (6) Support structure.

    (7) Design of all assemblies has been to the same environmental specifications.

    (8) Since the crossed-H is an independent spacecraft, it is designed to operate in the thermal, light intensity, radiation, and vacuum environment of space. There are no constraints on the CSM in these area, once separated and in orbit.

    (9) The assemblies are not sensitive to variation of atmospheric pressure, from normal to vacuum, while stored.

  (10) When in storage, the crossed-H is protected by a polyethylene dust cover.

  (11) No unusual constraints.

  (12) Acceleration during shipping and handling (incl. shock and vibration) typical for space hardware.

  (13) Operational vibration and accelerations are Saturn V launch and boost loadings. Stored and operational noise environments are also Saturn V launch and boost conditions.

  (14) In accordance with MIL-STD-826.

6. ENVIRONMENT CONSTRAINTS

a. Tolerance Limits

Constraint \ Assembly	(1)	(2)	(3)	(4)	(5)	(6)
Thermal	+160°F					
Stored	−65°F	(7)	(7)	(7)	(7)	(7)
Operational	(8)	(8)	(8)	(8)	(8)	(8)
Atmospheric Pressure (Stored)	(9)	(9)	(9)	(9)	(9)	(9)
Relative Humidity (Stored)	50%	50%	50%	50%	50%	50%
Air Movement Rate (Stored)	(10)	(10)	(10)	(10)	(10)	(10)
Atmospheric Composition (Stored)	(11)	(11)	(11)	(11)	(11)	(11)
Contaminants (Stored)	(10)	(10)	(10)	(10)	(10)	(10)
Acceleration (Storage)	(12)	(12)	(12)	(12)	(12)	(12)
Positive						
Negative						
Transverse						
Acceleration (Operational)	(13)	(13)	(13)	(13)	(13)	(13)
Positive						
Negative						
Transverse						
Vibration (Storage)	(12)	(12)	(12)	(12)	(12)	(12)
Random						
Sinusoidal						
Vibration (Operational)	(13)	(13)	(13)	(13)	(13)	(13)
Random						
Sinusoidal						
Noise (Stored)	(13)	(13)	(13)	(13)	(13)	(13)
Light Tolerance	(8)	(8)	(8)	(8)	(8)	(8)
Intensity						
Wavelength						
Radiation Tolerance	(8)	(8)	(8)	(8)	(8)	(8)
RFI	(14)	(14)	(14)	(14)	(14)	(14)
EMI	(14)	(14)	(14)	(14)	(14)	(14)

b. Interference

Since the crossed-H is an independent spacecraft in orbit, potential interference is minimal. During prelaunch operations, EMI is restricted to meet MIL-STD-826, and during launch the spacecraft systems are inactive.

7. DATA MEASUREMENT REQUIREMENTS

It is assumed that the discussion of data requirements here is provided for determination of the data interface with the CSM, and not for the assignment of specific operating frequencies and bandwidths for interface with the supporting ground stations. Since the crossed-H does not rely on the CSM data systems for relay of information, there is no data interface with the CSM. Data interfaces between the two spacecraft bodies, and between the lower satellite body and the ground station are illustrated by the information flow chart, Figure II-9.

NASA FORM 1346   Jan 67   9
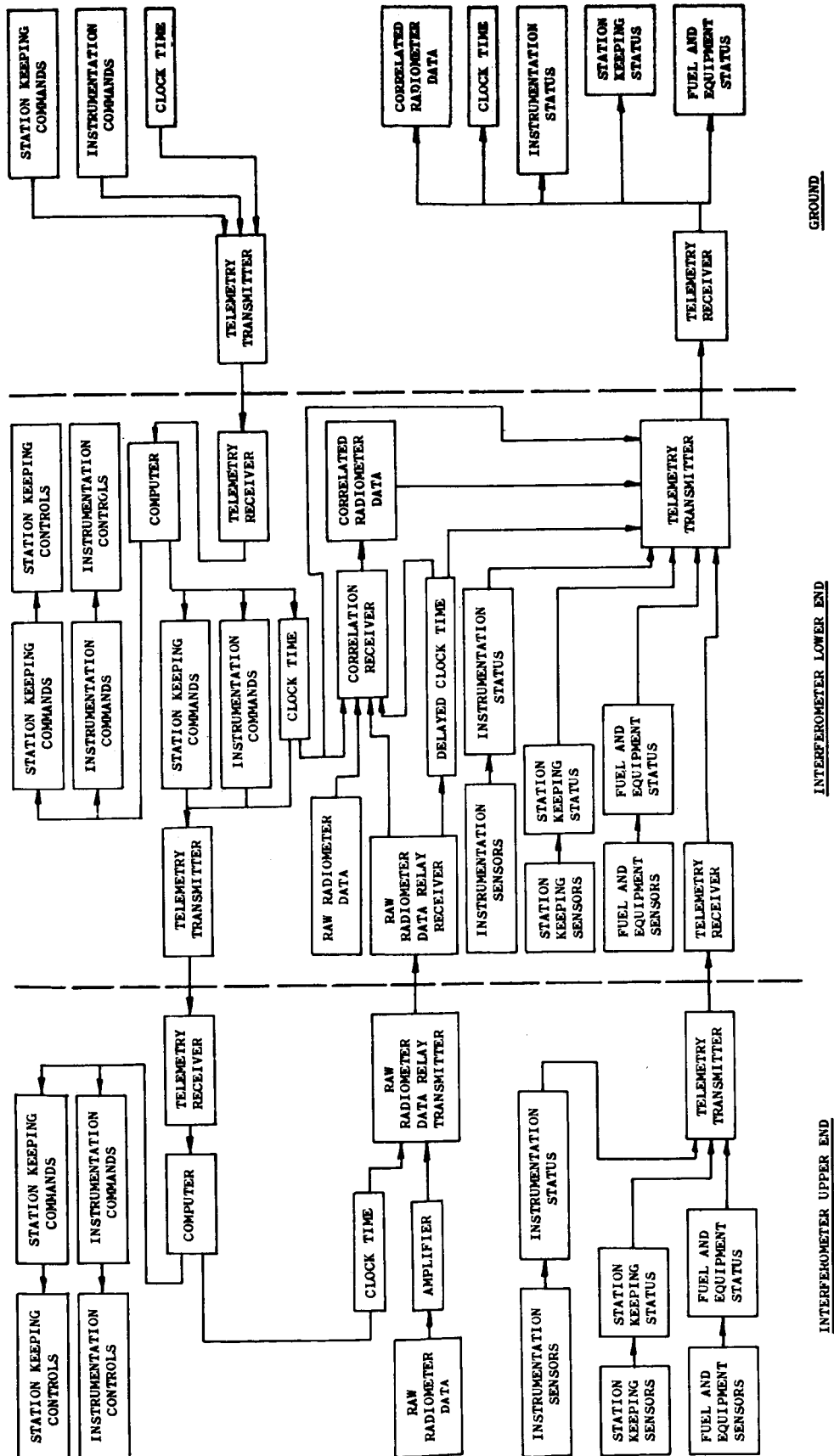
Figure II-9. Simplified Information Flow Chart

SECTION IV - OPERATIONAL REQUIREMENTS

1. SPACECRAFT ORIENTATION REQUIREMENTS

There are two primary spacecraft involved: the CSM and the crossed-H spacecraft. The interactions of these two vehicles, and the astronaut crew are critical to the successful conduct of the experiment.

a. Maneuvers

The CSM is used in the separation and deployment procedure. The deployment sequence is shown in Figure II-10.

b. Type of Orbit

The orbit most suitable for the objectives of the crossed-H is the circular synchronous orbit.

c. Orbit Parameters

The circular orbit has perigee and apogee altitudes of 10,200 n.mi., and a period of one sidereal day. Orbit inclination preferred is 28 1/2 deg; however, low inclination values are acceptable. Stationary position at the neutral-stable point near 120 deg W longitude is preferred.

d. Lighting Constraints

There is no requirement on the CSM or launch vehicle for adjustment of the attitude of the crossed-H with respect to the sun, and no other lighting constraints.

e. Launch Time

There is no specific launch window required. To meet secondary objectives, it may be preferable to launch at or near an equinox to provide shadow period while the CSM crew is available for observation of the eclipse effects on the large space structure, and for repair or identification of failures possibly induced by the shadowing.

f. Number of Measurements Required

Spacecraft (CSM) orientation requirements are not a function of the measurement program. Orientation of the crossed-H will be required throughout the measurement program, lasting a year or more.

g. Time Per Measurement

CSM orientation not required during the measurement program. The crossed-H measurement program requires the the times shown in Figure II-11.

NASA FORM 1346 Jan 67   11
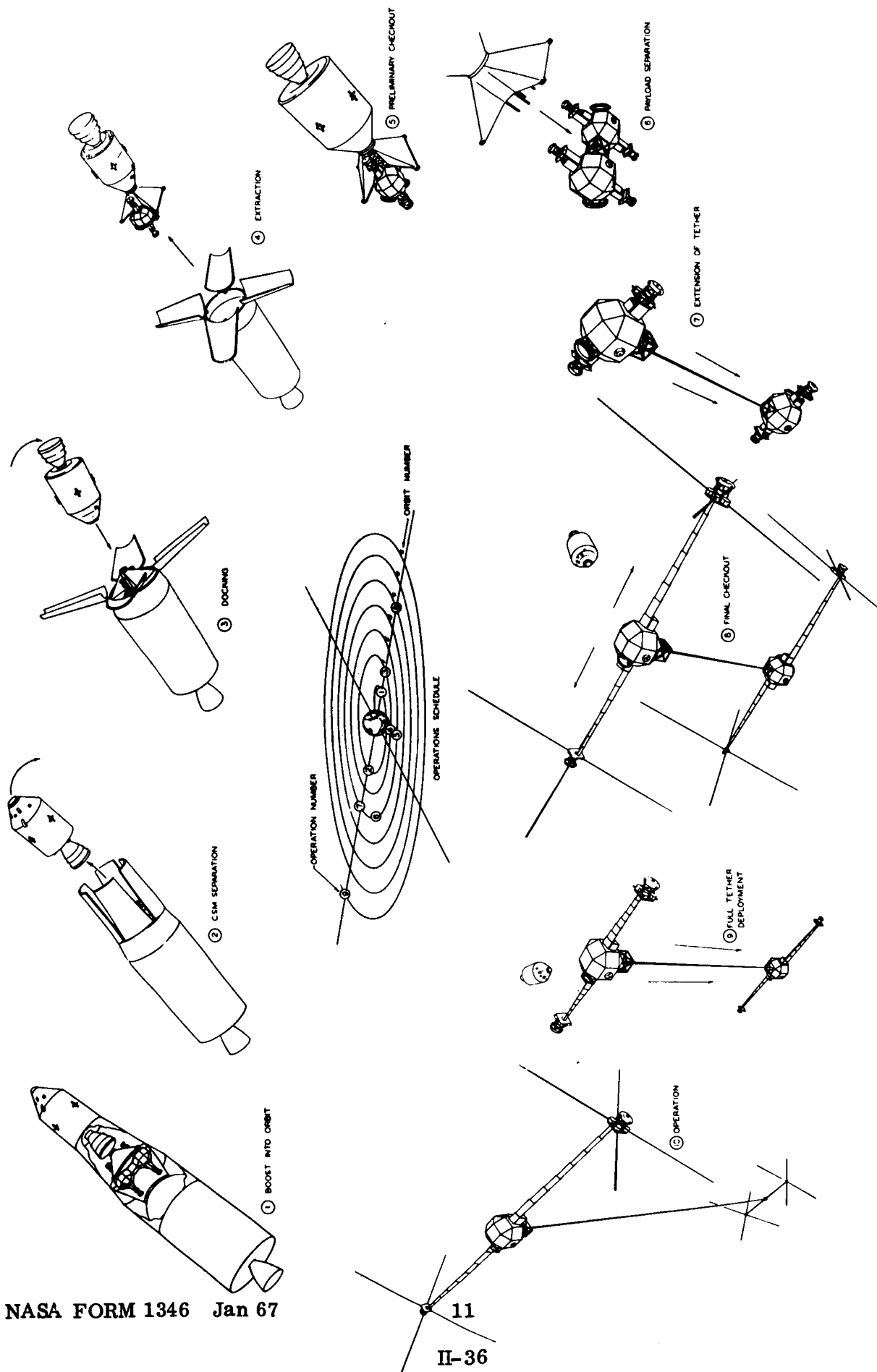
Figure II-10. Orbital Deployment Operations

OPTIONAL MODE

5.0 MHz TO 10.0 MHz

EXTEND TETHER TO 1,000 M LENGTH, POSITION AND STABILIZE IN GRAVITY GRADIENT ORIENTATION — 2.0 DAYS

REPEAT STEP 2 — 0.4 DAYS

SET DIPOLE LENGTHS AT 37.5 M AND DIPOLE SPACING AT 7.5 M. — 0.5 DAYS

REEL TETHER IN CONTINUOUSLY AT A RATE OF 21 M PER ORBIT UNTIL TETHER LENGTH IS REDUCED TO 62.5 M. — 44.5 DAYS

REPEAT STEPS 1 THRU 12 WITH BOOM AXES ORIENTED PARALLEL TO PLANE OF ORBIT AND 90° TO AXIS OF TETHER

2.5 MHz TO 5.0 MHz

EXTEND TETHER TO 2,000 M LENGTH, POSITION AND STABILIZE IN GRAVITY GRADIENT ORIENTATION — 2.0 DAYS

REPEAT STEP 2 — 0.4 DAYS

SET DIPOLE LENGTHS AT 75 M AND DIPOLE SPACING AT 15 M. — 0.5 DAYS

REEL TETHER IN CONTINUOUSLY AT A RATE OF 42 M PER ORBIT UNTIL TETHER LENGTH IS REDUCED TO 125 M. — 44.5 DAYS

REPEAT STEPS 1 THRU 8 WITH BOOM AXES ORIENTED PARALLEL TO PLANE OF ORBIT AND 90° TO AXIS OF TETHER

0.5 MHz TO 2.5 MHz

EXTEND TETHER TO FULL 10,000 M LENGTH, POSITION AND STABILIZE IN GRAVITY GRADIENT ORIENTATION. — 2.0 DAYS

ORIENT INTERFEROMETER ENDS WITH BOOM AXES 45° FROM PLANE OF ORBIT AND 90° FROM AXIS OF TETHER. — 0.4 DAYS

SET DIPOLE LENGTHS AT 150 M AND DIPOLE SPACING AT 30 M. — 0.5 DAYS

REEL TETHER IN CONTINUOUSLY AT A RATE OF 84 M PER ORBIT UNTIL TETHER IS REDUCED TO 250 M. — 116.1 DAYS

TOTAL MAPPING TIME
LESS OPTIONAL MODE = 332.8 DAYS

TOTAL MAPPING TIME
WITH OPTIONAL MODE = 427.6 DAYS

Figure II-11. Hypothetical Source Survey Mission

h. Orbital Location During Measurements

Measurements are made throughout the orbit. Ground station/satellite geometry remains constant.

i. Spacecraft Pointing Accuracy

Requirements for the CSM are those of docking and maneuvering while docked to a compact passive satellite body. Pointing accuracy of the crossed-H interferometer is ±1.0 degree along the antenna axis. Roll stability is not a critical parameter.

j. Allowable Spacecraft Rate

Drift rate during measurements is less than 1 degree per 4 minutes.

2. ASTRONAUT TRAINING

Refresher Course in
Radio Astronomy
Theory

Equipment
Usage

Malfunction Detection,
Analysis & Repair
Techniques

Safety Procedures

Task Simulation
1. Radio Astronomy Tasks:
   a. Source detection/scanning.
   b. Discrete source investigation.
   c. Alignment.
   d. Calibration.
   e. Data analysis.
2. EVA Structural Tasks:
   a. Detection and repair of malfunction in deployment.
   b. Adjustment of interlocks.
   c. Electrical connections.
   d. Installation and change of components.
   e. Inspection.
   f. Maintenance.
   g. Tool handling.

Mission Simulation and Familiarization.
Sequential steps in Mission Procedure
for deployment and initial
operation of the Crossed-H
Interferometer

Figure II-12. Summary of Astronaut Training Program

3. ASTRONAUT PARTICIPATION PLAN

Activities of the astronauts during the experiment are as follows:

a. Experiment Separation

The sequence followed in the separation of the crossed-H interferometer is that generally proposed for large space structures; the astronaut participation plan is based on the sequence for docking to and separating the LEM from the adapter. All three astronauts participate in the separation of the CSM from the launch vehicle. It is maneuvered to dock to the support structure of the crossed-H interferometer. On command from the CM, the crossed-H interferometer payload is separated from the launch vehicle and pulled to a safe distance for deployment.

b. Experiment Deployment

Pre-deployment subsystem checks are performed by the crew, and photographs taken of the undeployed experiment. On command from the CM, the crossed-H interferometer satellites are separated from the support structure and the CSM backs away with the support structure (and its complement of other experiment payloads). After additional checks and photography of the crossed-H interferometer, the experiment is deployed to its first operational configuration, by command from the CM.

c. Post Deployment Equipment Checkout

Initial operation of the crossed-H interferometer will be monitored from the CSM, and telemetered data processed and relayed to the CSM for correlation with visual observations.

d. Failure Correction

Provisions will be made for replacement of malfunctioning modules in the following sample areas:

(1) Solar cells

(2) Actuation motors

(3) Dipole units

(4) ACS modules

(5) Batteries

The activities by event are detailed in the following table.

FIRST DAY IN ORBIT					
EVENT	TASK AND EQUIPMENT	ASTRONAUT A	B	C	TIME (MIN.)
1.	PERFORM PRE-EVA.	X	X		120
2.	CABIN DEPRESSURIZATION.	X			4
3.	OPEN HATCH AND EXIT CM.	X	X		5
4.	MANEUVER TO STOWAGE AREA, ANCHOR TO STRUCTURE, UNSTOW CLOTHES LINE RIG AND ATTACH IT TO STRUCTURE.	X	X		8
5.	RELEASE ANCHORS AND MANEUVER (WITH ONE END OF THE CLOTHESLINE) TO SOLAR CELL PANELS ON SATELLITE NO. 1.	X			5
6.	ANCHOR TO A SUITABLE STRUCTURE AND ATTACH CLOTHESLINE TO STRUCTURE.	X			8
7.	ATTACH EIGHT SOLAR CELL PANELS ON THE CLOTHESLINE (FOUR TRIANGLE AND FOUR LARGE PANELS).		X		5
8.	REST.	X	X		2
9.	TRANSFER A LARGE SOLAR CELL PANEL TO ASTRONAUT A AT THE WORKSITE.	X	X		5
10.	INSTALL ONE LARGE SOLAR CELL PANEL ON SATELLITE NO. 1.	X			3
11.	TRANSFER A TRIANGLE PANEL TO ASTRONAUT A AT WORK SITE.	X	X		3
12.	INSTALL A TRIANGLE SOLAR CELL PANEL ON SATELLITE NO. 1.	X			3
13.	REST.	X			2
14.	RELEASE WORK SITE ANCHORS AND RELOCATE TO INSTALL FOUR ADDITIONAL PANELS ON SATELLITE NO. 1.	X			8
15.	REPEAT EVENTS (9) THROUGH (13) TWO MORE TIMES TO INSTALL FOUR PANELS ON SATELLITE NO. 1.	X			32

EVENT	TASK AND EQUIPMENT	ASTRONAUT A	B	C	TIME (MIN.)
FIRST DAY IN ORBIT (CONTINUED)					
16.	RELEASE WORK SITE ANCHORS AND ONE END OF CLOTHESLINE AND MANEUVER ON SATELLITE NO. 2.	X			8
17.	ANCHOR TO A SUITABLE STRUCTURE ON THE WORKSITE AND ATTACH CLOTHESLINE TO STRUCTURE.	X			7
18.	REPEAT EVENTS (9) THROUGH (13) ONE TIME TO INSTALL TWO SOLAR CELL PANELS ON SATELLITE NO. 2.	X			16
19.	RELEASE WORK SITE ANCHORS, REMOVE CLOTHESLINE RIG, RETURN TO STOWAGE AREA AND ANCHOR.	X			10
20.	REST.	X			2
21.	REMOVE CLOTHESLINE RIG AND STOW.	X	X		4
22.	RELEASE ANCHORS AT SM AND RETURN TO CM.	X	X		5
23.	ENTER THE CM AND CLOSE HATCH.	X	X		10
24.	REPRESSURIZE THE CM AND REMOVE THE EVA EQUIPMENT.	X	X		65
SECOND DAY IN ORBIT					
25.	PERFORM PRE-EVA.	X	X		120
26.	CABIN DEPRESSURIZATION.	X			4
27.	OPEN HATCH AND EXIT CM.	X	X		10
28.	MANEUVER TO STOWAGE AREA, ANCHOR TO STRUCTURE, UNSTOW CLOTHESLINE RIG AND ATTACH ONE END TO STRUCTURE.	X	X		8
29.	RELEASE ANCHORS AND MANEUVER (WITH ONE END OF CLOTHESLINE) TO DIPOLE MODULES ON SATELLITE NO. 1.	X			5
30.	ANCHOR TO A SUITABLE STRUCTURE AND ATTACH CLOTHESLINE TO STRUCTURE.	X			7
31.	ATTACH SIX DIPOLE MODULES ON THE CLOTHESLINE.			X	5

NASA FORM 1346   Jan 67   12

SECOND DAY IN ORBIT (CONTINUED)				
EVENT	TASK AND EQUIPMENT	ASTRONAUT A B C		TIME (MIN.)
32.	TRANSFER A DIPOLE MODULE TO ASTRONAUT A AT WORK SITE.	X X		5
33.	FLIP EXPENDED MODULE TO STOWAGE POSITION AND INSTALL NEW MODULE FOR SAT. 1.	X		3
34.	REST.	X X		2
35.	REPEAT EVENTS (32) THROUGH (34) THREE MORE TIMES FOR SATELLITE NO. 1.	X X		30
36.	RELEASE WORK SITE ANCHORS AND CLOTHESLINE AND MANEUVER TO DIPOLE MODULES ON SATELLITE NO. 2.	X		10
37.	ANCHOR TO SUITABLE STRUCTURE ON THE WORK SITE AND ATTACH CLOTHESLINE TO STRUCTURE.	X		6
38.	REPEAT EVENTS (32) THROUGH (34) TWO MORE TIMES FOR SATELLITE NO. 2.	X X		20
39.	RELEASE WORK SITE ANCHORS AND CLOTHESLINE, MANEUVER TO SM AND ANCHOR.	X		10
40.	REMOVE CLOTHESLINE RIG (ATTACHED TO SM STRUCTURE) AND STOW IN STOWAGE AREA.	X X		4
41.	RELEASE ANCHORS AT SM AND RETURN TO CM.	X X		5
42.	ENTER THE CM AND CLOSE HATCH.	X X		10
43.	REPRESSURIZE THE CM AND REMOVE THE EVA EQUIPMENT.	X X		65
THIRD DAY IN ORBIT				
44.	PERFORM PRE-EVA.	X X		120
45.	CABIN DEPRESSURIZATION.	X		4
46.	OPEN HATCH AND EXIT CM.	X X		10
47.	MANEUVER TO THE SM AND ANCHOR TO STRUCTURE.	X X		7
48.	UNSTOW CLOTHESLINE RIG AND ATTACH ONE END TO STRUCTURE ON SM.	X X		4
49.	RELEASE ANCHORS AND MANEUVER (WITH ONE END OF CLOTHESLINE) TO AN ACS TANK.	X		4

EVENT	TASK AND EQUIPMENT	ASTRONAUT A	B	C	TIME (MIN.)
	THIRD DAY IN ORBIT (CONTINUED)				
50.	ANCHOR TO THE WORK SITE AND ATTACH CLOTHESLINE TO STRUCTURE.	X			4
51.	UNSTOW FUEL HOSE FOR ACS TANKS FROM STOWAGE AREA (CONCURRENT WITH EVENT 50).		X		4
52.	ATTACH HOSE NOZZLE TO CLOTHESLINE AND TRANSFER TO ASTRONAUT A AT WORK SITE.		X		4
53.	REST.	X	X		2
54.	REMOVE FUEL HOSE NOZZLE FROM CLOTHESLINE AND ATTACH TO ACS TANK USING TOGGLE LATCH.	X			4
55.	REMOVE WORK SITE ANCHORS, MANEUVER TO A SAFE DISTANCE AND ANCHOR.	X			5
56.	OPEN VALVE IN CM TO START REFUELING.		X		1
57.	AFTER COMPLETION OF REFUELING, TURN OFF FUEL AND DEPRESSURIZE HOSE (FROM CM).		X		4
58.	REMOVE ANCHORS (FROM SAFE AREA) AND MANEUVER TO ACS TANK (JUST REFUELED).	X			4
59.	UNLATCH AND REMOVE HOSE FROM ACS TANK AND ATTACH TO CLOTHESLINE.	X			4
60.	REST.	X			2
61.	REMOVE CLOTHESLINE, RELEASE ANCHORS, AND MANEUVER TO NEXT ACS TANK.	X			6
62.	ANCHOR TO WORK SITE AND ATTACH CLOTHESLINE TO STRUCTURE.	X			4
63.	REPEAT FOR REMAINING ACS TANKS.	X		X	140
64.	REMOVE CLOTHESLINE AND ANCHORS.	X			4
65.	MANEUVER TO SM AND ANCHOR TO STRUCTURE	X			4
66.	REMOVE CLOTHESLINE AND STOW IN SM.	X	X		4
67.	RELEASE ANCHORS AT SM AND RETURN TO CM.	X	X		5
68.	ENTER THE CM AND CLOSE HATCH.	X	X		8
69.	REPRESSURIZE CM AND REMOVE EVA EQPT.	X	X		65
70.	CHECKOUT EXPERIMENT.	X	X	X	50

END OF MANNED SUPPORT ACTIVITIES.

4. PRE-LAUNCH SUPPORT

a. Preliminary Shipping and Handling Procedures

Material handling and packaging is controlled by National Aerospace Standards (NAS). These standards establish the methods to be used during the handling and shipping phases of the program.

Handling and shipping fixtures are used for movement of the crossed-H during in-plant movement and for shipment of the experiment in its packaged configuration. Preparation for delivery instructions provide protection against damage and degradation during shipment to the destination. For final delivery of the crossed-H, the handling and shipping fixture for the final assembly is secured to a skidded base suitable for handling by crane or fork-lift truck. It is shrouded with a barrier material to exclude dirt or other foreign contaminants. Container sides, ends, and top are provided to protect against impact damage. Explosive hardware, solar cell arrays, and batteries are packaged and shipped in separate containers. Containers are marked in accordance with MIL-STD-129, including hazardous warnings and shipping piece numbers, to ensure proper handling and ready identification at the destination.

b. Preliminary Installation and Checkout Procedures

The crossed-H and ground support equipment is shipped to MSFC for MSFN network compatibility tests and Saturn-V fit checks.

(1) MSFN Compatibility

The spacecraft is removed from the shipping container, inspected, and subjected to standardized checkout of electrical, command, telemetry and experiment subsystems. Test equipment, procedures, and operations are identical to those which will be used at the launch site.

(2) Saturn-V Fit-Check

The crossed-H, LEM adapter, and CSM are mated to determine their mechanical/structural compatibility.

Following these tests the spacecraft is repackaged for shipment to KSC/AFETR for launch operations.

Shipping and receiving of all crossed-H equipment at the launch site are accomplished per shipping instructions listing all deliverable items, with inspection and storage instructions for each.

Engineering confidence checks are then performed on all spacecraft subsystems to verify proper operation after shipment and prior to interface checks. Two engineering procedures are performed: solar array electrical test and electrical storage battery charge.

Interface checks are then performed in all operating modes for both flight and backup subsystems.

Upon completion of interface checks the spacecraft is assembled to prepare for mating with its payload support assembly. The procedure includes:

(1) Cleaning of spacecraft/support assembly interfaces.

(2) Installation of flight batteries and solar panels.

(3) Verification of proper mating of spacecraft electrical connectors.

A systems test is then performed which duplicates the spacecraft systems acceptance test performed at the factory. The payload is then installed in the LEM adapter and mounted atop the Saturn launch vehicle.

The flight readiness checkout procedure includes four sections:

(1) Spacecraft system checks.

(2) Experiment system checks.

(3) Installation, checkout, and interface verification.

(4) Final spacecraft flight preparation.

c. Facilities

Launch operations for the Saturn-V vehicle, conducted at Complex 39, utilize the mobile, or off-pad-assembly, concept. This provides for greater flexibility and launch rate than on-pad assembly, and employs four basic operations:

(1) Vertical assembly and checkout of the Saturn-V and all payloads on a mobile launcher in a controlled environment.

(2) Transfer of the assembled and checked-out vehicle to the launch pad on a mobile launcher.

(3) Automatic checkout at the launch pad.

(4) Launch operations by remote control from the laumch control center.

The major units involved are the vertical assembly building, mobile launcher, launch control center, mobile service structure, crawler transporter, launch pad, and high pressure gas facility.

d. Test Equipment

Test equipment used in prelaunch operations is standard laboratory equipment available in the vertical assembly building. The launch control panel installed in the launch control center is the only required piece of operational ground equipment.

e. Services

Compressed nitrogen (4000 psi) is required for fueling the crossed-H attitude control system.

---

5. FLIGHT OPERATIONAL REQUIREMENTS

The crossed-H spacecraft is an independent spacecraft operating at synchronous altitude. Initial operations are manned and partially controlled in orbit. Subsequent operations are unmanned and automated for remote control by ground stations, as follows:

a. Command Control

b. Telemetry Data Acquisition

c. Data Reduction and Evaluation

The onboard subsystems are designed to be compatible with the MSFN and DSIF stations.

Detail design of the subsystems will require further inputs from the principal investigator(s).

---

6. RECOVERY REQUIREMENTS

Recovery of all or part of the X-HIE is not required.

7.   DATA SUPPORT REQUIREMENTS

(To be completed by the Principal Investigator.)

NASA FORM 1346   Jan 67   13

SECTION V - RESOURCE REQUIREMENTS

PHASE C AND D ONLY - CROSSED "H" INTERFEROMETER

1. FUNDING REQUIREMENTS

a. Summarize total experiment cost by major category of expenditure as outlined below:

ITEM	AMOUNT
DIRECT LABOR (Separate by Labor Category; Rate per hour or man-month; Personnel involved, what they will do, etc.)	$ 10,765
MANUFACTURING BURDEN (Overhead) RATE (%) (Flight experiments normally will be supported by contracts rather than grants.)	
MATERIALS (Total) (Bill of Material, including estimated cost of each major item.)	9,925
SUBCONTRACTS (List those over $25,000) (Specify the vendor if possible, and the basis for estimated cost. Include baseline study contracts.)	7,145
SPECIAL EQUIPMENT (Total) (List of lab equipment, purposed uses, and estimated costs.)	95
TRAVEL (Estimated number of individual trips, destinations, and costs.)	70
ANY OTHER ITEMS (Total) (Explain in detail similar to the above.)	
TOTAL COSTS	$ 28,000
*General and Administrative Rate ( )	$
TOTAL ESTIMATED COST	$

b. Funding Obligation Plan. Provide the preliminary funding requirements of the experiment by quarter as indicated on the attached sheet (Quarterly Funding Requirements). Funding should be broken into the general areas indicated on the following page and should identify the source of funding for each area.

*Included in Direct Labor.

CROSSED "H" ANTENNA (PHASE C AND D ONLY)

SECTION V - RESOURCE REQUIREMENTS (Cont'd)
C. Quarterly Funding Requirements (Dollars in Thousands)

ITEMS	FUNDING SOURCE	FY 1 Q1	FY 1 Q2	FY 1 Q3	FY 1 Q4	FY 2 Q1	FY 2 Q2	FY 2 Q3	FY 2 Q4	FY 3 Q1	FY 3 Q2	FY 3 Q3	FY 3 Q4	FY 4 Q1	FY 4 Q2	FY 4 Q3	FY 4 Q4	TOTALS
Definition, Breadboard, Design, Development, Fabrication, Test (mock-ups, prototypes, and support equipment)		515	1090	2045	3125	3565	3145	2560	1620	940	290	110	85	25				
Fabrication, Test and Delivery (Flight units and spares)							700	1065	1385	1420	1155	660	430	25				
Supporting Studies and Other Implementation Efforts		160	160	160	160	165	170	170	170	170	170	155	155	80				
Data Analysis and Publication																		
YEARLY TOTALS			7,415				14,715				5,740				130			28,000

GRAND TOTAL

SECTION V - RESOURCE REQUIREMENTS (Cont'd)
2. PRELIMINARY DEVELOPMENT SCHEDULE

PLANNED DEVELOPMENT SCHEDULE

MAJOR MILESTONES	FY 70 Q1	FY 70 Q2	FY 70 Q3	FY 70 Q4	FY 71 Q1	FY 71 Q2	FY 71 Q3	FY 71 Q4	FY 72 Q1	FY 72 Q2	FY 72 Q3	FY 72 Q4
EIP COMPLETE – 2nd Qtr FY 69 (1347)												
MSFEB Action – 3rd Qtr FY 69												
Hardware Contract	◁											
ICD Complete				◁								
Design Complete						◁						
DEP Complete						◁						
Prototype Delivered									◁---◁			
Qualification Testing Complete										◁		
Flight Units Fabricated											◁	
Delivery of Flight Hardware												◁

SECTION V - RESOURCE REQUIREMENTS (Cont'd)

3. MANPOWER

Provide a brief summary of manpower requirements both in-house and contract.

---

4. FACILITIES

Provide a brief listing of facilities and major lab equipment requirements. Specifically identify new facility requirements. Whenever possible, indicate the schedule of usage for each item.